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XC-142A V/STOL TRANSPORT TRI-SERVICE LIMITED CATEGORY I EVALUATION

ROBIN K. RANSONE
Chief Test Engineer

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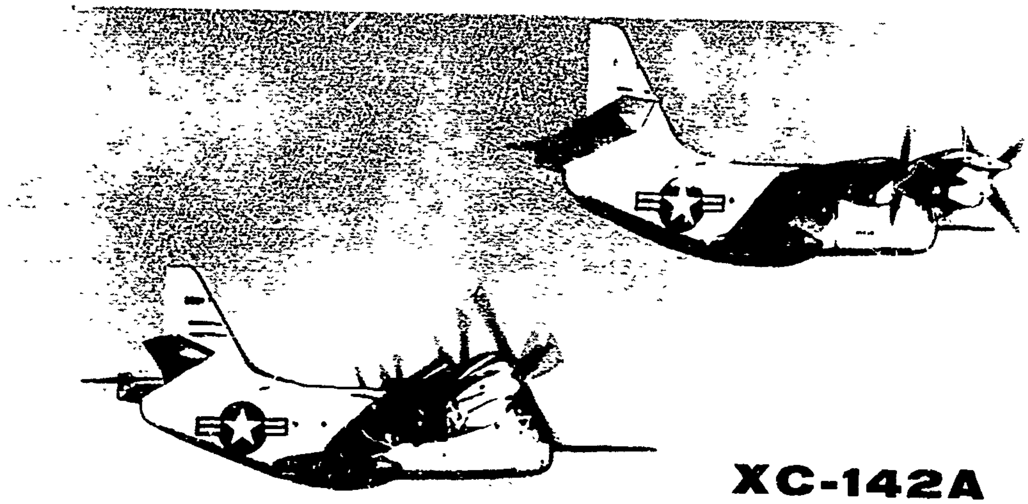
TECHNICAL REPORT No. 65-27
JANUARY 1966

AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE

FTC-TR-65-27



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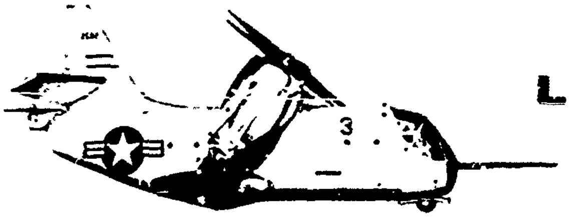

ROBIN K. RANSONE
Chief Test Engineer

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Major, USAF
Chief Test Pilot

FOREWORD

The XC-142A V/STOL Transport Tri-Service Limited Category I evaluation was conducted at the Ling-Temco-Vought facility in Dallas, Texas, by the Tri-Service V/STOL Test Force from the Air Force Flight Test Center (AFFTC), Edwards Air Force Base, California. The tests were conducted from 17 March to 29 April 1965 under the authority of Program Structure 478A and Project Directive 62-45.

This report contains no classified information extracted from other classified documents. Foreign announcement and dissemination by the Defense Documentation Center is not authorized because of technology restrictions of the U. S. Export Control Acts as implemented by AFR 400-10.

The Field Test Force organization was as follows:

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Robert Fickling, Captain, USAF, AFFTC/FTDT

The aircraft and instrumentation were serviced and maintained by the contractor during these tests.

This technical report has been reviewed and is approved.

15 December 1965

Albert M. Cate

ALBERT M. CATE
Colonel, USAF
Deputy for Systems Test

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ABSTRACT

A 9-hour 52-minute limited performance, stability and control, and systems evaluation of the XC-142A V/STOL transport was made at the contractor's Dallas, Texas, facility by the Tri-Service V/STOL Test Force from Edwards Air Force Base, California. The tests were made to define the aircraft's capabilities, problems, and to determine its readiness for Category II testing. Stability Augmentation System (SAS) ON V/STOL operations were safe, straightforward, and easy for the pilot to accomplish; SAS OFF required more pilot attention and proficiency, but was acceptable for emergency operation. The thrust in hover was about 12 percent less than predicted, which limited the VTOL and hover performance. A conversion or reconversion could be made when enough power was available to safely hover. STOL performance was not as good as predicted. At 41 300 pounds, the heaviest weight tested, takeoff distance over a 50-foot obstacle was 500 feet. Landing distance over a 50-foot obstacle was 860 feet, which was too long for an aircraft of this type and was not compatible with the short takeoff distances. At the lightest weight tested of 34 340 pounds, the XC-142A did not meet the short landing criteria of 500 feet total distance over a 50-foot obstacle. Cruise performance was about 11 percent less than predicted. The only potential tilt-wing concept limit was a severe stability and control deterioration near the ground at wing angles between 40 and 80 degrees. This did not restrict conversions or reconversions above 25 feet. Most of the deficiencies were with the systems which were not sufficiently tested before installation due to the limited funds available. The novel and critical systems (i.e., flight control system, gearboxes, cross-shafting, wing tilt, etc.) were more thoroughly tested before installation and gave little trouble. Three safety of flight deficiencies were found, plus many other deficiencies which would hamper the Category II tests if not corrected. With the correction of the "A" and "B" deficiencies listed in the Conclusions and Recommendations section of this report, the XC-142A would be ready for Category II. The XC-142A was not ready for production. Many deficiencies were identified which should be corrected for a production C-142 aircraft.

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LIST OF ABBREVIATIONS, SYMBOLS, AND DEFINITIONS

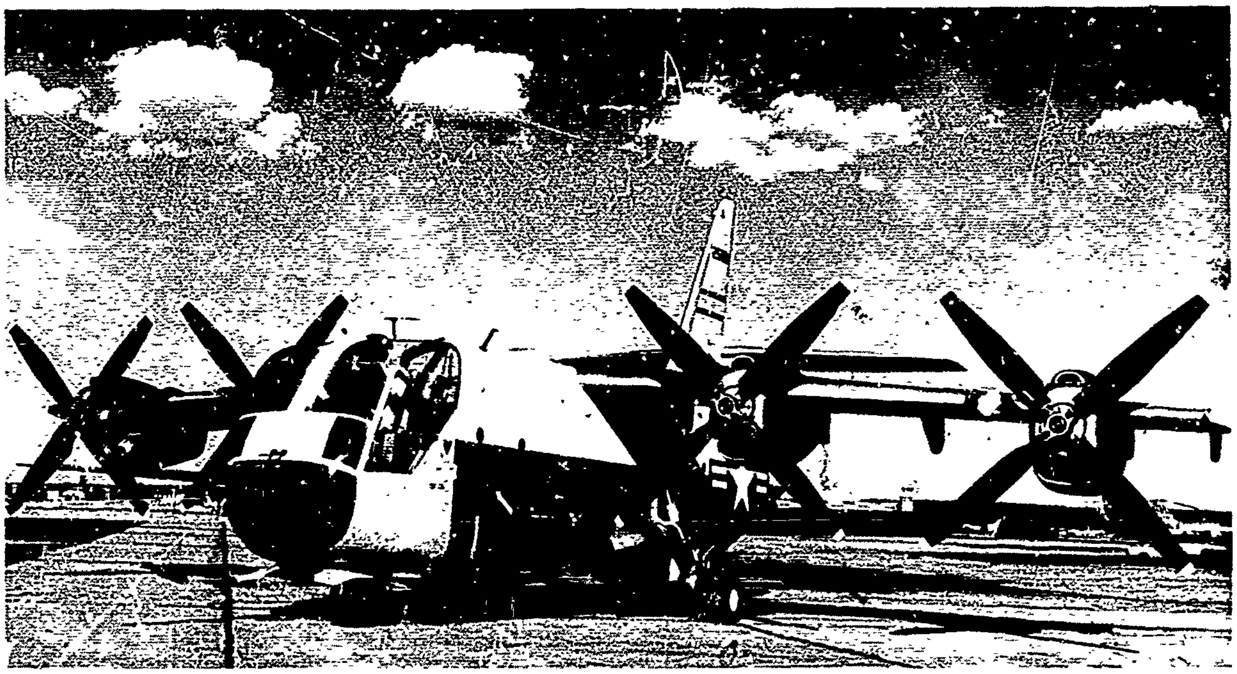
Symbol	Definition	Units
ac	alternating current	---
AFFTC	Air Force Flight Test Center	---
AFR	Air Force Regulation	---
AFSCM	Air Force Systems Command Manual	---
AGARD	Advisory Group for Aeronautical Research and Development	---
AGE	aerospace ground equipment	---
APU	auxiliary power unit	---
ARTO	air run takeoff - a vertical takeoff followed immediately by initial conversion at minimum useable terrain clearance height	---
ASG	Aeronautical Standards Group	---
beta	propeller blade pitch angle	---
Btu	British thermal unit	---
C	centigrade	---
CA	conversion angle - the angle measured from the longitudinal axis of the aircraft to some meaningful reference in the lift mechanism. This angle is approximately 90 degrees in hover and approximately zero degrees in cruise. (For the XC-142A a conversion angle of 0/30 degrees denotes a wing tilt angle of zero degrees and a flap deflection of 30 degrees.)	---
CAS	calibrated airspeed	knots
cg	center of gravity	percent MGC
Conversion Speed	The optimum airspeed for the aircraft to complete transition and enter conventional flight.	---
C_P	power coefficient	dimensionless
CR	Cooper Rating	---
CSD	constant speed drive	---

C_T	thrust coefficient	dimensionless
db	decibel	---
dc	direct current	---
Deadman's Zone	The combinations of airspeed and altitude near the ground at which an engine failure would cause the aircraft to contact the ground hard enough to cause structural damage.	---
deg	degree	---
dn	down	---
EO	engineering order	---
F	Fahrenheit	---
FM	frequency modulation	---
FOD	foreign object damage	---
fpm	feet per minute	---
fps	feet per second	---
FSN	Federal Stock Number	---
ft	feet	---
FTDT	Bioastronautics Branch, AFFTC	---
FTTED	Systems Engineering Division, AFFTC	---
FTTEE-4	Trainer and Miscellaneous Branch, Performance Engineering Division, AFFTC	---
FTTOB	Bomber-Transport Operations Division, AFFTC	---
FTTV	Tri-Service Test Force, AFFTC	---
fwd	forward	---
g	gravitational acceleration	32.2 feet per second per second
HF-SSB	high frequency-single side band	---
HSD	Hamilton Standard Division	---
IFF	identification friend or foe	---
IFR	instrument flight rules	---

IGC	integral gearcase	---
ILS	instrument landing system	---
KCAS	knots calibrated airspeed	knots
KEAS	knots equivalent airspeed	knots
KIAS	knots indicated airspeed	knots
kt	knot	---
KTAS	knots true airspeed	knots
kva	kilovolt-ampere	---
lat	lateral	---
lb	pound	---
long.	longitudinal	---
lt	left	---
LTV	Ling-Temco-Vought, Inc.	---
min	minute	---
MGC	mean geometric chord	---
Mod Spec	Model Specification	---
MPP	Military pilot participation in the contractor's Category I test	---
NAMPP	nautical air miles per pound of fuel	nautical miles per pound
NAMT	nautical air miles traveled	nautical miles
N_f	free turbine speed	rpm
N_g	gas generator speed	rpm
NM	nautical mile	nautical miles
N_p	propeller speed	rpm
OAT_s	standard day outside air temperature	degrees C
OAT_t	test day outside air temperature	degrees C
OB	outboard	---

PC-1	No. 1 power control system	---
PC-2	No. 2 power control system	---
pct	percent	---
PIO	pilot induced oscillation	---
PITS	propulsion integrated test stand	---
PLF	power for level flight	---
psi	pounds per square inch	---
rpm	revolutions per minute	---
rt	right	---
SAS	stability augmentation system	---
sec	second	---
shp	shaft horsepower	horsepower
shp_t	test day shaft horsepower	horsepower
shp_{corr}	test day shaft horsepower corrected for unstabilized conditions	horsepower
shp_{iw}	generalized shaft horsepower	horsepower
SL	sea level	---
S/N	serial number	---
SPO	Systems Program Office	---
STOL	short takeoff and landing	---
STO/VC	short takeoff with vertical climbout - the operational technique of using a short ground run before takeoff to alleviate ground effect problems and a vertical climb to clear a 50 foot obstacle.	---
t.e.d.	trailing edge down	---
t.e.u.	trailing edge up	---
T.O.	technical order	---
T_{t_5}	power turbine inlet temperature	degrees C
T/W	thrust to weight ratio	dimensionless

UHF	ultra high frequency		---
UHT	unit horizontal tail		---
USA	United States Army		---
USAF	United States Air Force		---
USN	United States Navy		---
V_c	calibrated airspeed	knots	
Verti-circuit	The maneuver which consists of a vertical takeoff, transition, retransition, and a vertical landing.		---
VFR	visual flight rules		---
VHF	very high frequency		---
V_i	indicated airspeed	knots	
V_{ic}	indicated airspeed corrected for instrument error	knots	
V_{iw}	generalized airspeed	knots	
V_{max}	maximum airspeed	knots	
VOR/LOC	VHF omnirange/localizer ILS Receiver		---
ΔV_{pc}	airspeed position error correction	knots	
V_t	true airspeed	knots	
VTOL	vertical takeoff and landing		---
W_{f_t}	test day fuel flow	lb per hr	
$W_{f_{corr}}$	test day fuel flow corrected for unstabilized conditions	lb per hr	
W_{iw}	standard weight for generalized parameters	pounds	
W_t	test day gross weight	pounds	



INTRODUCTION

GENERAL

The objectives of these tests were to provide an early military evaluation of the XC-142A vertical or short takeoff and landing (V/STOL) transport aircraft's capabilities, characteristics, and problems; and to determine its readiness for Category II testing.

The tests were made at the Ling-Temco-Vought (LTV) facility, Dallas, Texas, from 17 March to 29 April 1965, by a Tri-Service test team of engineers and pilots. There were 19 flights during this period for a total flight time of 9 hours 52 minutes, of which 7 flights were made with an all-military crew and 12 flights were with a combined military-contractor crew. The military pilots had also monitored and participated in contractor flights from 12 January 1965.

The heaviest weight tested was 41 500 pounds. Most of the military tests were made at a mid center of gravity (cg).

Systems testing of the XC-142A was restricted to that which could be accomplished in conjunction with the performance and stability and control tests. Many systems were not evaluated to their design limits or maximum ranges, and failure simulation tests were not accomplished because operating envelope limits had not yet been demonstrated by the contractor.

The systems evaluation con-considered the following: (1) peculiarities of the various systems, (2) problem areas which were outstanding at the time of these tests, (3) problem areas which were discovered and corrected on

the XC-142A and should be considered on future V/STOL aircraft, (4) a limited maintenance evaluation, and (5) design features which should be considered if a C-142 aircraft is produced.

Persistent systems problems contributed to a high percentage of maintenance downtime and a high abort rate. A complete reliability program was not pursued in the design of the XC-142A due to the austere nature of the program.

AIRCRAFT DESCRIPTION

The XC-142A was a 4-engine tilt-wing V/STOL transport designed to demonstrate the visual flight rules (VFR) and instrument flight rules (IFR) operational capabilities of the tilt-wing concept. Normal crew consisted of a pilot (right seat) and copilot (left seat), although the aircraft was designed for safe operation by one pilot if necessary. Zero altitude-zero speed rocket-type ejection seats were provided for the pilot and copilot. The XC-142A was designed to carry 32 fully-equipped combat troops, 24 litter patients, or 8000 pounds of cargo. The aircraft had its own auxiliary power unit (APU) for hydraulic and electrical power and required no ground support equipment.

The XC-142A was powered by four General Electric T64-GE-1 free turbine turboprop engines, each rated at 2850 shp, mounted in nacelles on the wing. The engines drove four variable-pitch fiberglass 4-bladed tractor propellers and a 3-bladed fiberglass tail propeller through an interconnected gear and shaft train. Thus, power was available to turn all five propellers when one, two, or three engines were shut down. The tail propeller rotated in a horizontal plane and was declutched and braked for cruise flight.

The major aircraft dimensions are shown in figure 1. The XC-142A was about three-fourths the size of a C-123. The design empty weight was 23 926 pounds, the design VTOL weight was 37 474 pounds, and the design overload (STOL) weight was 43 700 pounds; design cg limits were 15-percent mean geometric chord (MGC) to 28-percent MGC (reference 3). Integral fuselage tanks provided 1400 gallons of usable fuel. Ferry tanks were not provided on these aircraft. The wing had an 8.53 aspect ratio, and was mounted high on the fuselage. Except for the center portion the wing was completely immersed in the propeller slipstream for stall suppression during transition and low-speed flight. The wing could be tilted through an incidence angle of 98 degrees by two screw-jack actuators driven by a centrally located hydraulic motor. Wing tilt was controlled by a variable rate switch on each collective lever or by a constant rate switch on each No. 4 throttle. The wing incorporated full span double-slotted trailing edge flaps in three sections, with the center and outboard sections operated also as ailerons. The flaps were programed automatically with wing tilt, although the pilot had an override capability. Leading edge slats for stall suppression were mounted outboard of each engine nacelle and operated automatically as a function of flap position. The vertical tail consisted of a conventional fin and rudder arrangement and supported the slab-type unit horizontal tail (UHT) assembly.

The fully-powered irreversible flight control system had artificial feel forces and was powered by dual independent hydraulic systems. Dual cockpit controls were provided; these consisted of conventional rudder pedals and control sticks, plus collective levers for all takeoffs and landings. The collective levers were essentially pro-

MAJOR AIRCRAFT DIMENSIONS

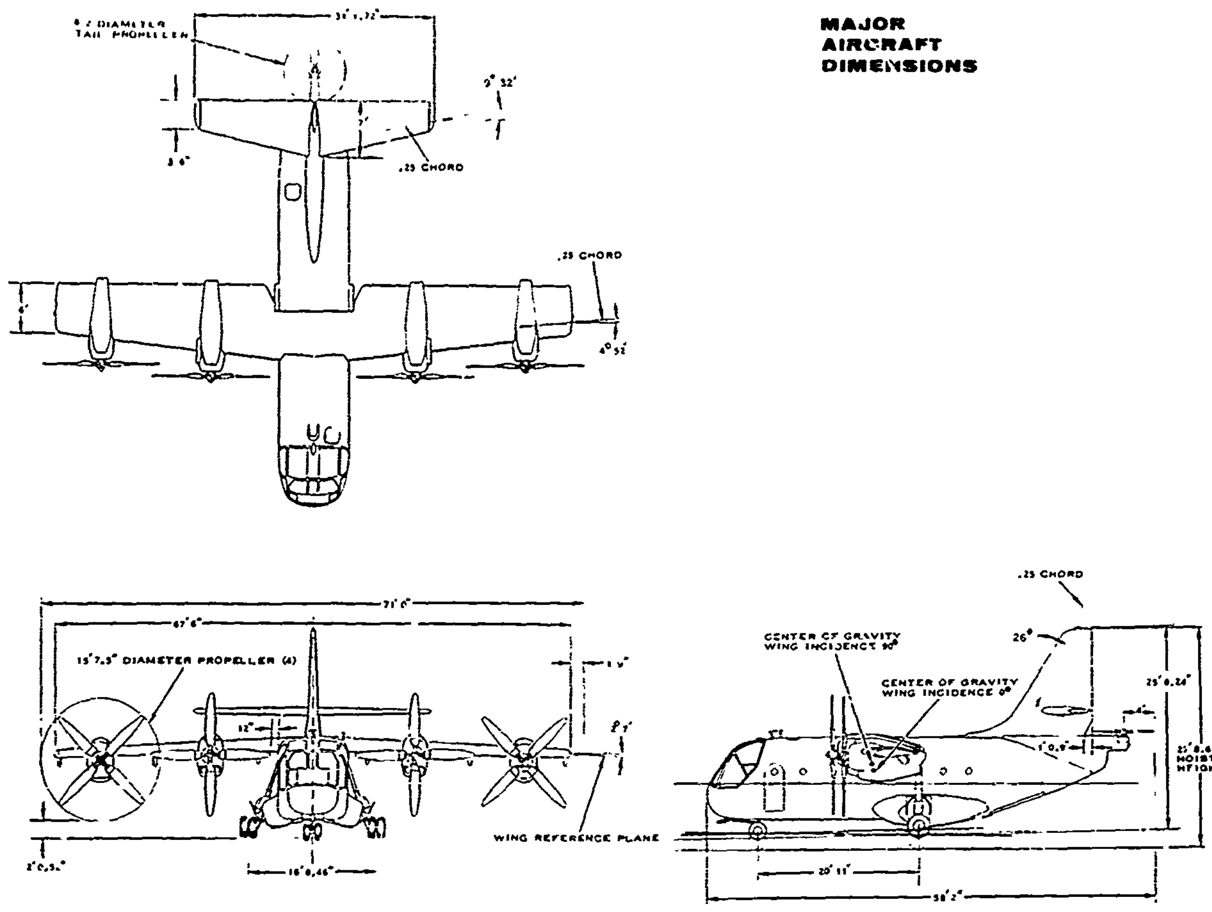


figure 1

peller blade pitch angle (beta) controls but were also connected to the throttles to control engine power. Trim was provided about the pitch, roll, and yaw axes for all modes of flight and was achieved through trimming the primary control surfaces. In cruise flight the cockpit controls operated the rudder, UHT, and ailerons. During VTOL operation these same cockpit controls operated the ailerons (yaw), tail propeller beta (pitch), and differential propeller beta between the right and left wing (roll). Control function phasing was automatically accomplished by a completely mechanical integrator package which was referenced to wing tilt angle. Thus, the primary flight control to be used for each axis varied with the mode of flight (i.e., wing tilt angle). At intermediate wing angles, two flight control surfaces could function together to achieve a net control about one axis.

A stability augmentation system (SAS) provided rate and attitude damping in roll and pitch, and rate damping in yaw and height. It was used only in the VTOL (hover) and STOL (transition) flight regimes. All stabilization gains, except for pitch rate damping, were a function of wing angle and were phased out for cruise. Full pitch rate damping was available any time the tail propeller switch was in the ENGAGE position. The rate

damping system in roll, pitch, and yaw was a dual comparative system which automatically drove to neutral if the two channels disagreed. Single channel operation was possible.

TEST AND EVALUATION

GROUND OPERATIONS

TOWING

Towing the aircraft presented no problems. There were no turning limits on the nose gear since it could be rotated 360 degrees while moving the aircraft. A long or short universal MD-1 tow bar could be used; however, the long bar was mandatory for clearance while towing aircraft equipped with the test nose boom. An F-102 tow bar with an adapter could also be used.

ENTRY AND LOADING

Normal crew entrance was by a door on the left forward portion of the fuselage. A 3-step stand was usually available at the factory to reach the 40-inch-high cargo compartment personnel entrance. There were no current plans for self-contained, portable, or integral steps with onboard storage. Suitable steps should be provided on production aircraft. (D 36)[1]

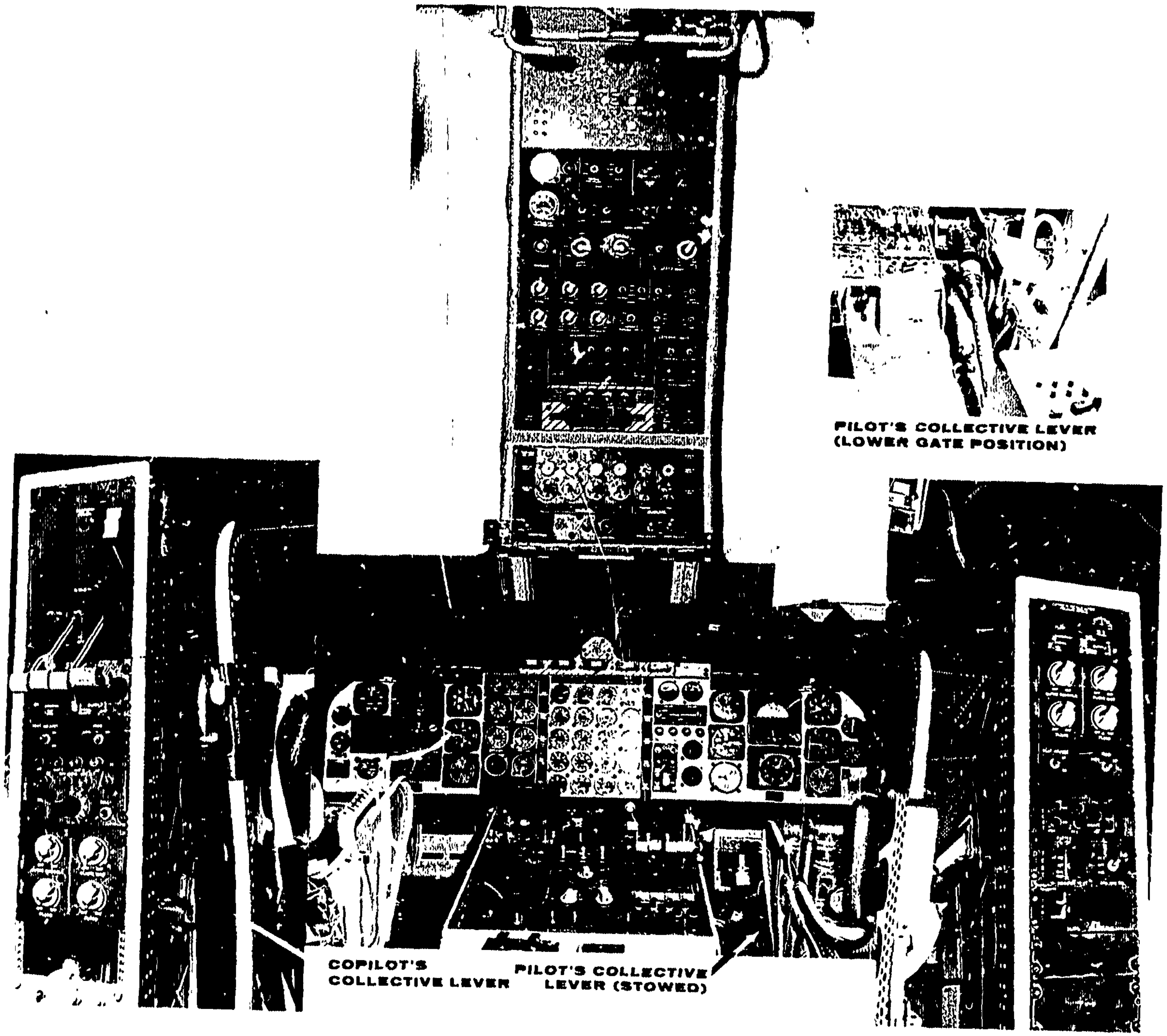
Cargo or troops were to be handled through rear doors which operated much like those on the C-130 to provide a 7-foot-high by 7 1/2-foot-wide entrance to the 7- by 7 1/2- by 30-foot cargo compartment. These doors had not been cleared to be opened in flight. They should be operated in flight at the maximum allowable speed and with the tail propeller operating before the end of Category I testing. (C 9)

Crash landing-ditching exit hatches were located in the top of the fuselage above each pilot and at the forward left and aft right corners of the cargo compartment. Ladders were provided to reach the overhead hatches in the cargo compartment. An alternate bailout hatch was located in the forward left cargo compartment floor. Pins to prevent the overhead hatches in the cargo compartment from vibrating loose in flight would have made egress under emergency conditions doubtful since the pins had to be pulled and the handle actuated to release the hatches. This condition violated Air Force Systems Command Manual (AFSCM) 80-1 (C.6-2.4.3.3) (reference 6) and was a flight safety deficiency which should be corrected as soon as possible. (A 1)

COCKPIT EVALUATION

Two steps we. . provided to gain access to the flight deck from the cargo compartment. The pilot and copilot were supplied with zero-zero rocket type ejection seats which were not scheduled to be installed in production models. Zero-zero ejection seats should be provided in production aircraft. Normal pilot ejection was to be through the overhead plexiglass panels. Entry and egress from the seats was awkward and could not be accomplished without bumping into cockpit switches or controls. The seat had an electrically-actuated adjustment for vertical movement, but no fore and aft adjustment

[1] Numbers indicated as (D 36)., etc. represent the corresponding recommendation numbers as tabulated in the Conclusions and Recommendation section of this report.



PILOT'S COLLECTIVE LEVER
(LOWER GATE POSITION)

was possible to bring the pilot to a comfortable position in relation to the aircraft controls. Fore and aft adjustment should be provided in production aircraft as recommended in AFSCM 80-1, Volume III, drawing AD3. The rudder pedal adjustment was adequate and rapid and was controlled by means of a twist knob on the rudder pedal pedestal between the rudders. (D 52, D 53)

Except for the vertical direction, ground and air visibility from the cockpit was excellent for all modes of flight. Vertical visibility was restricted because the overhead plexiglass had been painted to minimize the "greenhouse" heat buildup. Overhead visibility was desirable because of the aircraft vertical takeoff capability and should be provided on production models. It was possible that glare and reflection problems would exist at night due to the large curved plexiglass areas. (D 34)

A swingout jump seat for the flight mechanic was located between and slightly aft of the two pilots. The jump seat was not evaluated during the tests.

A total torque instrument, not provided on the test aircraft, was planned for the Category II aircraft instrument panel. This was not necessary since individual engine torque indicators were provided. The wing-flap position indicator had been moved to the top of the instrument panel glare shield and a flight test instrument was substituted for the production instrument on the aircraft commander's panel. The production indicator had wing position markings every 10 degrees but had numerals every 40 degrees. This made it extremely difficult to correctly position the wing. (D 43, B 39)

The standby compass was located on a bracket in the center windshield area. The compass blocked an unnecessary amount of forward visibility and should be moved to a less prominent position. (D 42)

The landing gear handle was located at the lower left position on the right instrument panel and was not accessible to the copilot. It should be relocated or duplicated to permit actuation by either pilot while his shoulder harness is locked. (D 51)

The engine instruments were grouped in the center panel in a normal multiengine fashion. The engine oil pressure gages were at the bottom of the center panel, but the gearbox oil pressure and gearbox and engine oil temperature gages were on the overhead panel and were difficult to read because of the mounting angle and aft location. Pilots considered it desirable to have dual needle gearbox-engine oil pressure gages and similar dual needle temperature gages located on the center panel. This arrangement would conform with AFSCM 80-1 (C.2-1.3.3.1) and alleviate the problem of reading the overhead temperature and pressure instruments. The emergency fire T-handles were at the top of the center panel. The APU T-handle was located in the center between the T-handles for engines No. 2 and 3. The APU handle should be changed to allow immediate physical distinction from the engine fire handles. (C 40, D 49)

The center pedestal contained communications and navigation equipment, SAS controls, caution annunciator panel, and engine controls for the right seat pilot. Duplicate throttles and rpm control levers were located on the left outboard pedestal for the copilot. The propeller governor switch was located on the extreme right side of the center pedestal beyond the copilot's reach. It should be re-

located or dualized to permit actuation by either pilot with his shoulder harness locked. The tail propeller controls were located on the center console and were not acceptable because the actuation of the controls was subject to misinterpretation or actual misuse. Misuse could cause failure and/or severe damage to the tail propeller drive system. This item has been under study by the contractor. A simple ON-OFF control for tail propeller engagement and disengagement should be provided for production aircraft and should be evaluated during Category I. (C 47, C 69)

The emergency hydraulic system switch and the emergency wing control switches were located side by side on the forward portion of the overhead panel, while the primary-secondary wing control switch was located on the center pedestal. This created confusion which could be eliminated by grouping the above switches on a wing control panel. (C 46)

The separate declutch and feather controls on the overhead panels were undesirable since both controls had to be actuated in proper order, first the declutch, then the feather control, to feather a propeller. This two-step procedure created a critical time lapse during the feathering operation. Pilots desired the separate declutch and feather controls, but with the addition of an automatic declutch in the feather control. This would provide automatic declutch and feather capability with the actuation of a single control while maintaining the separate declutch action needed in propeller trouble analysis. (D 79)

The two-sectioned overhead panel was located too far aft and too nearly horizontal for good visibility. The overhead panel should be inclined more and shifted forward. The Chip-detector wafer switch was located on the aft portion of the overhead panel and could not be reached without releasing the safety belt and shoulder harness. It should be relocated for easier access and to conform to AFSCM 80-1 (C.2-1.3.1). (D 44, D 50)

A green-jewel-type oil cooler door position light illuminated when the number two oil cooler door was open. A more desirable indication for the oil cooler doors would be a warning of an out-of-sequence condition. (D 84)

The battery's only function was to start the aircraft APU. No electrical power was available on the aircraft for intercommunication, emergency signals, or lights until the APU was started. Emergency lighting, intercom, and alarm system power should be provided by the battery as prescribed in AFSCM 80-1 (C.6-2.4.1.4.4). (D 74)

Green lights, located on the overhead panel, indicated proper engine oil service levels, but were deactivated when the engine switches were turned on. No indication of low oil levels was available during flight. The addition of oil and hydraulic fluid quantity indicators or low level warning devices in the cockpit is desirable. (D 62, D 82)

The gearbox oil pressure caution light remained on with a propeller feathered. This prevented monitoring the remaining operative gearbox oil pressures and should be changed. (D 76)

The hydraulic system orientation was not satisfactory. This problem is discussed in the Hydraulic System section.

The wing caution light was on when the utility hydraulic control handle was in the ON/WING UNLOCKED

position. This resulted in the caution light being illuminated for all takeoffs and landings. It should be modified to eliminate the warning light illumination during normal operating conditions. (C 75)

No provisions had been made for ashtrays, map cases, letdown chart holders, pilot footrests, thermos jug holders, cup holders, and other crew comforts. Though these items were eliminated by contractor and Systems Program Office (SPO) agreement to reduce development costs, they should be provided in production aircraft. (D 110)

The aircraft was equipped with conventional stick and rudder controls and a helicopter-like collective power lever which was used for all takeoffs and landings. The collective lever head contained a stability augmentation system control switch, a wing tilt control switch, and a flap override switch which were all thumb-operated. The sense of t.. ap override switch was confusing to some pilots in that an upward thumb movement extended and a downward movement retracted them. The collective lever also had a finger-actuated collective throttle release switch. It was used during landings to get below the flight idle collective gate, and in cruise flight to disconnect the throttle and stow the collective lever. During STOL landings it was necessary to raise the collective slightly to actuate the switch if the gate was reached without having first pressed the release. This delayed attaining ground idle power and increased the landing ground roll. A simple detent or increased friction force should be provided instead of the gate. (B 45)

A selective "hot mike" feature, similar to the AIC-18, should be provided for interphone communication. The majority of the navigation equipment could not be evaluated since it had been removed to make room for test instrumentation. Communications problems are discussed in the Avionics section. (C 94, D 95)

No provisions were made for simulating instrument flight conditions. Some equipment will be needed for an IFR evaluation during Category II and should be provided. (C 54)

ENGINE STARTING

GENERAL

Engine starting procedures for the XC-142A were good. The start was accomplished by placing the engine master switch ON, activating the crank and ignition relays, and placing the throttle in ground idle. Start power was provided by an APU in the right main wheel well. The APU provided hydraulic power for engine cranking and electrical power for the main ac and dc buses. Battery power was available only for starting the APU. At 62-percent gas generator speed (N_g), a signal transmitted from the engine tachometer terminated the automatic start functions. At this point, the starter pump was returned to pumping operation and provided hydraulic power to the related aircraft power control (PC) system. The starting circuit was so arranged that the APU could be used to start only one engine on a side. Parallel use of the operating engine hydraulic pump was used in starting adjacent engines. Normal starts required from 35 to 70 seconds, which was too long. The engine starting times should be reduced. (D 57)

ENGINE RUNUP

All four propellers turned when the first engine was started. The area forward of the propellers

and under the wing in the space where the propellers and wing rotate had to be kept clear.

Difficulties were experienced in keeping the aircraft stationary during engine operation above 30-percent torque. The parking brakes held but the engine power could skid the aircraft across the surface and cause flat spots on the tires. This was caused by the high thrust available in relation to aircraft weight and will be a problem common to this type of V/STOL aircraft. The aircraft was tied down with cables during high power ground checkout operations.

TAXIING

The lowest recommended wing-tilt angle for taxi was 10 degrees. Small bank angles could cause the outboard propellers to strike the ground at lower wing angles. At zero wing angle, the outboard propeller ground clearance was only 25 inches. Taxiing was initiated by releasing the parking brake and slightly raising the collective lever. Speed was controlled by brakes and collective lever. Adequate power was available for taxiing with the engines operating at the idle setting of 65-percent propeller rpm. Braking was provided through hydraulic boost pressure from the utility hydraulic system which was modulated through the toe brakes. The brakes faded on several occasions during downwind or down slope operation and were unsatisfactory. The brakes are further discussed in the Short Landing and Hydraulic System sections. The braking should be improved. (C 22)

Directional control during taxi was provided by an electrically-controlled hydraulically-actuated nosewheel which was steerable through 132 degrees. The nosewheel steering operated from rudder pedal movements and was alternately engaged or disengaged by pressing a switch on the control stick. A green light located on the lower left of the pilot's panel which indicated nosewheel steering ENGAGED was unnecessary. The circuit should be changed to provide nosewheel steering only while the switch is pressed. Additional directional control was provided at normal taxi wing angles (10 to 15 degrees) by differential main propeller beta obtained through the rudder pedal deflections. The nosewheel steering and differential propeller beta combined to provide excellent directional control during ground operation. At 10 degrees of wing angle differential propeller blade angle without nosewheel steering provided satisfactory directional control for all taxi operations except tight turns. (C 89)

PERFORMANCE AND HANDLING QUALITIES TESTS

VERTICAL TAKEOFF AND CONVERSION

Vertical takeoffs were made with the wing tilted 85 to 90 degrees. Longitudinal trim requirements were reduced by tilting the wing slightly off the vertical before takeoff to hover over a spot under the given wind conditions. In the VTOL configuration, longitudinal (pitch) control was achieved through the control stick by varying the tail propeller beta; lateral (roll) control was achieved through the control stick by differentially varying main propeller beta; and directional (yaw) control was achieved through the rudder pedals by differentially deflecting the ailerons which were in the propeller slipstream. The collective lever was essentially a main propeller beta and engine power control and was used to control height through power and thrust.

With smooth collective application the aircraft became airborne with no apparent instabilities or abrupt changes in power required. The height control was good when sufficient excess engine power was available and was given a Cooper Rating of 2 (CR 2) (see table XVII, appendix I). A discussion of power available in the vertical mode is in the Hover section.

The climb into hover was smooth, positive, and stable, and required only minor control motions in stable air to easily maintain a precise position over the ground (CR 2). Good control was present in pitch, roll, and yaw. Stick and collective forces during hover and conversion were good (CR 2) but the rudder forces were too heavy and were not harmonious with the other control forces (CR 4). These forces were much too high for these configurations and should be reduced by 50 percent and reevaluated. The directional control force gradients are presented in figure 14, appendix I. This plot also shows that the VTOL, STOL, and hover configuration directional force gradients were about 4 times the maximum recommended by AGARD 408 and MIL-H-8501A. (C 15)

After lift-off the aircraft was either stabilized at some altitude for hover or was climbed to approximately 50 feet for conversion to forward flight. The conversion maneuver was very easy to accomplish and could be initiated from a hover or climb (CR 2). To make the conversion, the wing was lowered by actuating the wing control switch either incrementally or at a constant rate; the latter method resulted in a much smoother conversion. The angle of climbout could be controlled by management of power and wing rate. A slight reduction in power was necessary to make a level conversion. Once the wing travel was started, the aircraft was maintained in a fuselage level attitude with negligible trim changes and acceleration was good. With constant maximum rate of wing movement, the acceleration from hover to approximately 140 knots was achieved in less than 12 seconds. When a conversion was started from a hover with no increase in power the aircraft immediately started climbing. This indicated that a conversion could be made whenever there was sufficient power to safely hover. This is also discussed in the Hover section. A converstion time history is shown in figure 27, appendix I.

Two variations of the vertical takeoff are the air run takeoff (ARTO) and the short takeoff with vertical climbout (STO/VC). The ARTO is desirable to avoid the deadman's zone, and the STO/VC is desirable to minimize recirculation and ingestion of debris into the engines. Both maneuvers require partial conversions very close to the ground. Neither of these tests was done during this evaluation due to the severe deterioration of handling qualities close to the ground at wing angles between 40 and 80 degrees. This problem is discussed in detail in the Short Landing section.

The procedure for configuring the XC-142A for climb or cruise was more involved than for most aircraft. Landing gear retraction during conversion resulted in no noticeable trim change since it was done below 40 KIAS. As the wing was lowered to zero degrees the tail propeller was dephased and braked by placing the clutch lock lever in the OFF position and placing the tail propeller switch in the BRAKE position. The collective was stowed by either pilot, and power control reverted to the throttles. Stowing the collective lever required the following steps:

1. Disengage the collective lever from the throttles by pressing the throttle disconnect button on the collective grip.

2. Slowly raise the collective lever to the upper stop to position the beta linkage while the propeller governor maintains the propeller rpm.

3. Tighten the friction on the collective lever grip to enable the beta linkage to maintain position.

4. Disengage the collective lever from the beta linkage by pressing the beta latch switch on the collective grip.

5. Lower the collective lever to the stowed position in a detent on the deck of the cockpit.

While most conversions were made at a mid cg of 20-percent MGC, some were made at 19 and 17 percent. No deterioration in flying qualities was noted by the pilots and there was no significant increase in the amount of control deflection required.

HOVER

Hover was easily accomplished from a vertical takeoff by adjusting the collective lever for the power required to maintain a constant height. No change in power required to hover within ground effect was detected.

In calm air, a stable hover could be obtained at all heights except between 4 and 10 feet where there was an apparent gusiiness of the air possibly caused by some recirculation effects. The aircraft was still easily maintained in a position over the ground but an increase in control activity was noted. The hover control and attitude limits should be determined. The aircraft controllability and maneuverability about all axes were good (CR 2). In gusting winds control activity increased, especially about the roll axis. The aircraft had a strong weathercocking tenden-y. (C 12)

A comparison of hover directional controllability specification requirements with test results is presented in table I. Requirements for yaw angular displacement 1 second after control application were computed using the ferry mission gross weight of 44 500 pounds as specified in the XC-142A model specification (reference 1). The tests were actually conducted at an average gross weight of 33 920 pounds with all SAS on.

TABLE I

HOVER DIRECTIONAL RESPONSE

Reference	Response for Full Control Input (deg in first sec)	Response for First Inch of Control Displacement (deg in first sec)
MIL-H-8501A LTV Mod. spec	9.24	3.39
AGARD Rpt No. 408 (All Weather Op)	10.0	1.8
Test results	Left 5.0 Right 5.0	2.3 1.5

The XC-142A did not meet the specification requirements for yaw control, but pilot comments indicated the directional control response was satisfactory. The test results presented in figure 15, appendix I, show that directional controllability was greater to the left than to the right; however, the 4- to 5-knot wind could have accounted for this difference. The pilots noticed no difference (CR 3). Ground static control cycles indicated that full right aileron displacement was 44 degrees trailing edge down and 46 degrees trailing edge up versus ± 50 degrees predicted by reference 3. The left aileron was not instrumented during these tests.

During this evaluation, the thrust available to hover even with the altitude damper off was marginal at gross weights of about 35 000 pounds at a density altitude of 2000 feet with four engines operating. The engines produced less power than expected due to inadequate topping, and the propeller efficiency was lower than predicted. The Flight Manual predicted a hover ceiling of 9400 feet for this weight with four engines operating at maximum power.

Estimates based on test data indicated that the aircraft would not meet the detail specification requirements for hover ceiling, or vertical thrust to weight ratio at the VTOL design weight of 37 474 pounds. The hover ceilings determined are only approximate due to insufficient data, but the limited data indicated a thrust loss compared to the Flight Manual of about 12 percent. Since engine power appeared normal during light weight hover tests, this loss was attributed primarily to a loss in propeller efficiency. The hover performance is summarized in table II and also presented in figure 1, appendix I. The propeller efficiency should be improved and the engine topping procedures changed. This is also discussed in the Propulsion section. (C 14)

TABLE II

FOUR-ENGINE HOVER CEILING PERFORMANCE

Thrust to Weight Ratio (T/W) = 1.0

Standard Day

Hover Ceiling (ft)	Gross Weight (lb) Flight Manual	Gross Weight (lb) C_P-C_T Method (extrapolated)
Sea Level	44 500	38 900
2 000	42 900	37 100
4 000	41 000	35 300
6 000	39 000	33 500
10 000	34 200	29 900

For most of the hover tests the altitude damper was turned off since it absorbed 4 to 5 percent of available power. No degradation was observed in height stability and the control was improved with the altitude damper off. The damper was installed mainly for IFR hover and may be helpful for that condition.

Forward and rearward wing-fixed translations were done by two methods. The lift vector was changed by tilting the aircraft with longitudinal control either while holding constant power or while holding constant altitude and increasing power. Both methods changed the attitude of the aircraft proportional to the control input. Translation while holding constant power resulted in a gradual loss of altitude. Air taxi was accomplished fore and aft by decreasing or increasing wing angle respectively which resulted in fuselage level attitudes. The translations using longitudinal control indicated positive stick position and stick force stability, while the air taxi had no effect on either.

The maximum rearward speed attained was 20 knots, measured from a pace automobile with an anemometer. This was the maximum rearward speed attainable with 98 degrees of wing angle and a level fuselage attitude. Test results are shown in figure 17, appendix I.

Lateral translations were accomplished by banking in the direction of translation. The bank angle was not excessive or uncomfortable. Although the aircraft had good static directional stability while hovering in winds, this decreased to practically neutral static directional stability in sideward flight. Almost neutral rudder was required for the sideward flight. The maximum speed attained was 25 knots. Test results in sideward flight are shown in figure 16, appendix I. (C 11)

SHORT TAKEOFFS

For a short takeoff, the aircraft was taxied into position on the runway, the brakes were held, and the wing was set to the desired angle. The tail propeller and 30 degrees of flaps were used for all short takeoffs. The collective lever was raised to apply the maximum torque possible without skidding the tires (25- to 35-percent torque). At this power setting, lateral control compensated for any crosswind since the ailerons were already effective. Takeoffs were made with up to 15 knots of crosswind with no difficulty, and this preroll correction was near that required for lift-off with wings level. The brakes were released to prevent skidding and takeoff power was rapidly applied. The average time required for the engines to reach a peak torque of about 80 percent was 2.4 seconds, and the aircraft lifted off about 1.4 seconds later (i.e., brake release to lift-off averaged 3.8 seconds). For takeoffs at all wing angles, the longitudinal trim was set to zero and a level fuselage attitude was held during the acceleration, takeoff, and climbout. When safely airborne, the landing gear was retracted, and the conversion angle (CA) was lowered to 0/30 degrees[2] as soon as possible. The tail propeller was shut down and braked and the collective lever stowage was started as the aircraft accelerated through 100 KIAS at 0/30 degrees CA. After the tail propeller and collective lever were secured, the flaps were raised to zero. At this point the aircraft had accelerated to about 140 KIAS.

[2] A CA of 0/30 degrees denotes a wing tilt angle of zero degrees and a flap deflection of 30 degrees.

In the cruise configuration (gear and flaps up, wing down and locked, and at power for level unaccelerating flight) the XC-142A was in light aerodynamic buffet at 140 KIAS. This was designated the conversion speed.

The military pilots flew qualitative short takeoffs at several CA's from 10/30 to 45/30 degrees. Conversion angles of 35/30 were the highest possible without stability and control deterioration due to ground effect. Ground roll was less than 750 feet for all CA's but at 35 degrees of wing the ground distances were 175 feet or less as shown in table III. During this evaluation there was not time to adequately test more than one wing angle. Therefore, all quantitative short takeoff tests were made with four engines operating, a CA of 35/30 degrees, and a cg of 18-percent MGC at weights from 34 150 to 41 300 pounds. The maximum allowable takeoff weight was 41 500 pounds. The takeoff performance data are presented in figures 2 and 3, appendix I. A 35/30 degree CA takeoff time history is shown in figure 21, appendix I.

TABLE III
TAKEOFF PERFORMANCE COMPARISON
35/30 Degrees Conversion Angle

Weight (lb)	Ground Roll (ft) Flight Manual	Ground Roll (ft) Test	Takeoff Speed (KTAS) Flight Manual	Takeoff Speed (KTAS) Test	Total Distance Over 50-ft (ft) Flight Manual	Total Distance Over 50-ft (ft) Test	Speed at 50-ft Height (KTAS) Flight Manual	Speed at 50-ft Height (KTAS) Test
41 300	127	175	29	41	405	500	Not Available	51
37 474	75	125	23	38	290	404		50
34 150	48	97	18	36	234	358		49

NOTE: See figures 2 and 3 of appendix I for test conditions.
A takeoff time history is shown in figure 21, appendix I.

Fairchild Flight Analyzers were used to determine the takeoff times, distances, and ground speeds. The data were corrected only for wind, since power correction methods had not yet been determined.

The ground and air distances obtained during this evaluation were much greater than those which had been predicted by the Flight Manual. Some of this deviation could be attributed to the following: (1) only 30-percent torque could be applied prior to brake release, (2) propeller and power deficiencies, (3) the pilot proficiency in maximum STOL techniques, and (4) takeoff airspeeds were higher than predicted.

The short takeoff weight (for 500 feet total distance over a 50-foot obstacle) of the XC-142A was 41 300 pounds for a CA of 35/30 degrees, with four engines operating, and sea level standard day conditions.

In the pilots' opinion, the XC-142A took too long to return to trim after a directional disturbance in the STOL configuration, especially at wing angles above 35 degrees. This, coupled with high rudder forces, made precise directional control difficult. Lateral and longitudinal control forces and controllability were good, but overall harmony was poor due to the high rudder forces. The STOL static directional stability should be improved. (C 16)

Most of the evaluation was conducted at a cg of approximately 18-percent MGC but a brief qualitative evaluation was made at 17-, 19-, 24-, 26-, and 28-percent MGC. No deterioration in flying qualities was observed and there was no significant increase in the amount of control deflection required. The cg limits should be determined. (C 5)

On one of the flights with a combined military-contractor crew, an evaluation was made of the short takeoff characteristics of the XC-142A at wing angles greater than 35 degrees. Although it was planned to evaluate the aircraft at wing angles up to 60 degrees, the highest angle reached was 45 degrees. Previous flights indicated that the aircraft's stability and control deteriorated significantly at wing angles above 40 degrees.

The initial takeoff was made using 91-percent propeller rpm, 70-percent torque and a CA of 40/30 degrees. The power response was immediate. At lift-off aircraft directional excursions were greater than at wing angles below 35 degrees, but due to the rapid climb through 15 feet this was only momentary (CR 3).

The next takeoff was made at the same power setting with a CA of 45/30 degrees. Aircraft reaction following lift-off was similar to that experienced at the 40/30 degree CA except that the deterioration in directional control was more pronounced and lasted through a height of approximately 25 feet. The instability and deterioration in controllability at this CA was also apparent about the roll axis (CR 4).

These disturbances presented no serious control difficulty. The landing characteristics were worse and are discussed in the Short Landing section. Due to this deterioration of stability and control at higher wing angles, a CA of 35/30 degrees should be the highest used for short takeoffs until more tests can be made. Since this could be a tilt wing concept limit, further tests should be made and the STOL CA limits determined. (C 23)

■ CLIMBS

Maximum power climbs at 91-percent propeller rpm were made to the operational ceiling of 25 000 feet to evaluate the climb performance and aircraft handling characteristics. The XC-142A operational ceiling was limited to 25 000 feet by the lack of aircraft pressurization and several systems limitations. A summary of the climb performance is in table IV. The climb performance was outstanding for a cargo/transport aircraft, although it was about 1000 fpm less than predicted for 100-percent propeller rpm. At a gross weight of 35 900 pounds, the time to climb from sea level to 25 000 feet was 6.1 minutes. Aircraft acceleration following application

of maximum power was rapid. Brisk rotation to the climb attitude was required to avoid exceeding the climb speed. For the 4-engine climbs, the initial fuselage attitude was about 26 degrees noseup; however, this did not adversely affect forward visibility. The climb data are presented in figure 4, appendix I.

Lateral-directional control was good, but the aircraft was sensitive longitudinally and attention was required for airspeed control. Directionally the aircraft was insensitive to small sideslips. Even though the air was quite stable, persistent small yaw excursions of less than 2 degrees were observed. Trimmability in the climbs was good, although in the 4-engine climbs approximately 75 percent of the available yaw trim was required to relieve the pedal forces. This was probably a propeller rigging problem and was not representative.

Level flight accelerations were made at 1000, 4000, 8000, and 12 000 feet to check the maximum thrust variation with altitude and to determine the climb potential and best climb speeds. Two accelerations were made at each altitude from the buffet onset speed to about 250 KCAS. Although the test day temperatures were 13 to 16 degrees C hotter than standard, the data indicated the best climb speeds were about 180 KCAS at 1000 feet decreasing to 160 KCAS at 12 000 feet. The climb potential curve, as shown in figure 5, appendix I, showed that little benefit would be gained by climbing at a slower speed. The data indicated that the best climb speeds were at the buffet onset speeds for the conditions tested.

LEVEL FLIGHT

The 4-engine and 2-engine level flight performance of the XC-142A was evaluated by flying at airspeeds between buffet and the maximum speed as determined by power limit or the existing airspeed limit of 245 KEAS. The conditions tested are listed in table V.

The contractor's recommended propeller speeds were used for these tests. The minimum propeller speed of 75 percent could not always be achieved because of the adverse effect of cold soaking the propeller governor. In some cases, the minimum propeller speed attainable with the rpm control lever back against the stop was about 79 percent. The propeller governor should be modified so that the minimum 75-percent rpm can be attained in flight. (B 100)

The cruise data are summarized in table VI. These data were not corrected to standard temperatures.

One of the unusual features of the XC-142A was its ability to have one or more engines shut down and still operate all four propellers through the cross-shafting. This eliminated the drag of a windmilling or feathered propeller. The specific fuel consumption of the T64 engines decreased with increased power settings. As a result of these characteristics, the contractor recommended 2-engine cruise for maximum range. Both 2- and 4-engine data were obtained during this evaluation. Better range was obtained with two engines operating but the cruise speeds were lower than predicted. The power required data are shown in figure 10, appendix I.

The 2-engine cruise data were obtained at 6000, 10 000, and 20 000 feet. The air was too turbulent below 6000 feet to obtain sea level cruise data, and 25 000 feet was above the 2-engine absolute ceiling at 35 000 pounds gross weight.

TABLE IV

FOUR-ENGINE MAXIMUM POWER CLIMB PERFORMANCE SUMMARY

CA 0/0 Degrees, gear up 91-percent Propeller rpm

Sea Level Weight = 35 900 lb

Altitude (ft)	Time (min)	Rate of Climb (fpm)	Airspeed (KCAS)	Distance Traveled (NM)	Fuel Used (lb)
SL	0	7340	188	0	0
5 000	0.8	6230	179	2.3	62
10 000	1.6	5140	166	5.0	130
15 000	2.7	4100	157	8.5	203
20 000	4.1	3080	156	13.4	283
25 000	6.1	1970	156	22.0	382

Note: Data were not corrected to standard temperature. Test day temperature varied from 2.8 degrees C colder than standard at sea level to 4.4 degrees C hotter than standard at 25 000 feet.

TABLE V

SPEED POWER TEST CONDITIONS

Cruise Configuration cg = 21-percent MGC

Number of Engines Operating	Average Gross Weight (lb)	Average Pressure Altitude (ft)	Speed Range (KCAS)	$OAT_t - OAT_s$ (deg C)
4	34 750	10 000	157-231	+8
4	35 510	25 000	163-195	+2
2	35 060	6 000	145-228	+3
2	33 350	10 000	142-198	+7
2	33 830	20 000	145-178	+4

TABLE VI

CRUISE TEST PERFORMANCE SUMMARY

Cruise Configuration cg = 21-percent MGC

Number of Engines Operating	Gross Weight (lb)	Pressure Altitude (ft)	$OAT_t - OAT_s$ (deg C)	Standard Day Flight Manual KCAS	KTAS	NAMPP	Best Cruise KCAS	KTAS	NAMPP	Recommended Cruise KCAS	KTAS	NAMPP
4	34 750	10 000	+8	-	-	-	190	223	0.1040	198	231	0.1030
4	35 510	25 000	+2	-	-	-	172	254	0.1388	180	265	0.1375
2	35 060	6 000	+3	173	189	0.1290	175	192	0.1240	186	203	0.1227
2	33 350	10 000	+7	168	195	0.1435	175	205	0.1340	182	213	0.1326
2	33 850	20 000	+4	155	211	0.1695	156	216	0.1520	167	229	0.1505

Notes: 1. Data were corrected for dv/dt and dh/dt but were not corrected to standard temperature.

2. Recommended cruise was the highest speed at which 99 percent of maximum NAMPP was attained.

The highest recommended 2-engine cruise true airspeed occurred at 10 000 feet, although there was little difference in the true airspeeds at the three altitudes (203 KTAS at 6000 feet, 213 KTAS at 10 000 feet, and 209 KTAS at 20 000 feet).

The best specific range was 0.1520 nautical air mile per pound of fuel (NAMPP) at 216 KTAS at 20 000 feet. This was obtained with two engines operating. This speed was 3 KTAS lower and the range was 11 percent less than predicted in the Flight Manual. The recommended cruise speed was based on a higher speed to shorten the mission time with only a 1-percent range penalty. This recommended cruise speed was 229 KTAS and the range was 12 percent less than predicted in the Flight Manual.

The 2-engine cruise performance is presented in figures 6 through 8, appendix I.

The 4-engine cruise data were obtained simultaneously with the airspeed calibration at 10 000 and 25 000 feet altitude. There were no 4-engine cruise data in the Flight Manual for comparison to the test data since the recommended cruise was with two engines operating. The 4-engine data were obtained mainly for comparison with the 2-engine data. The highest recommended 4-engine cruise speed was 265 KTAS at 25 000 feet.

At 10 000 feet, the 4-engine best cruise speed of 223 KTAS was 9 percent higher than the best 2-engine cruise speed, but the specific range was 22.5 percent lower. The range at the 4-engine cruise speeds was better with two engines shut down than with four engines operating. At the best 4-engine cruise speed the 4-engine specific range was 19.5 percent less than with two engines operating and at the recommended 4-engine cruise speed the specific range was 17 percent less. At high altitude, the comparison was made with 4-engine data at 25 000 feet (the operational ceiling) and 2-engine data at 20 000 feet. The 4-engine best cruise speed of 254 KTAS was 17.5 percent higher than the best 2-engine cruise speed but the specific range was 9 percent lower. The 4-engine cruise performance is presented in figure 9, appendix I.

A plot of shp_{iw} versus V_{iw} is shown in figure 10, appendix I. It shows no noticeable difference in total power required between 2- and 4-engine cruise at the same conditions. This plot shows some characteristics at low speed that cannot be explained or verified with the limited data available.

The visibility in level flight was excellent (refer to the Cockpit Evaluation section).

DESCENTS

Descent boundary investigations were performed by the contractor to verify the predicted buffet onset speeds and maximum sink rates. Buffet onset speeds and maximum sink rates were demonstrated during military crew checkouts. Buffet onset and buildup characteristics were similar for all configurations observed. Buffet onset was very light but distinct. It intensified as the sink rate was increased by power reduction. The maximum sink rate was defined by moderate airframe buffet and a slight decrease in stability and controllability, primarily about the lateral axis.

The buffet onset speeds agreed closely with predictions, and the region of light buffet at higher sink rates was more than sufficient for pilot warning. This buffet region is discussed in greater

detail in the Stall Approaches section. The predicted descent boundary is shown in figure 2,

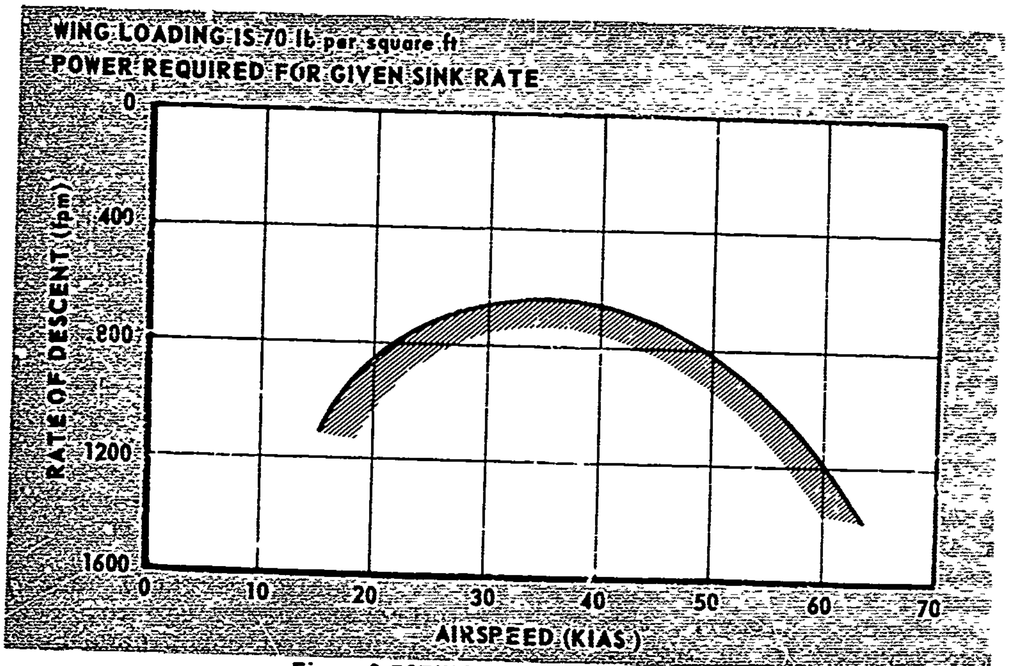


Figure 2 ESTIMATED DESCENT BOUNDARY

A phenomenon related to high-speed low-power clean configuration descents was experienced and was referred to as "loping." "Loping" was manifested in the cockpit by longitudinal acceleration pulses of small magnitude and varying frequency depending upon the speed-power configuration at the time. The lower the speed, the lower the pulsing frequency. The contractor attributed this phenomenon to the propellers operating at the minimum governing position and cycling between positive and negative thrust. This cycling of propeller thrust was transmitted to the aircraft and caused fore and aft wing movements. The movements were observed from the cockpit by both pilots and also by the chase pilots. The contractor considered this "loping" a high-priority study problem and recommended avoiding it during flight. The "loping" should be corrected. (D 19)

■ LANDING APPROACH

Setup of the landing configuration of the XC-142A was essentially the reverse of the cleanup procedure after takeoff. Although it was much more involved than for a conventional aircraft, there was sufficient time on the downwind leg. The following steps were required:

1. Decelerate to 150 KIAS.
2. Open the hydraulic isolation valve.
3. Select wing primary system.
4. Set 30 degrees of flaps.
5. Set 91 percent propeller rpm.
6. "Hook up"[3] the collective lever.

[3] The throttles moved with the collective lever when the collective lever was "hooked up."

7. Engage the tail propeller.

8. Lower the landing gear.

9. Raise the wing to 10 degrees.

10. Set final CA on short final approach.

The aircraft landing operation was very flexible with regard to approach speeds, glide angle, and pattern size. With the wing down, the pattern was similar to most fixed wing patterns. Higher wing angles permitted reduced pattern size and speed and allowed operations comparable with those of helicopters.

The downwind leg was normally flown at a CA of 10/30 degrees which resulted in a fuselage level trim speed of 92 KIAS. On base leg in the landing pattern, the flaps were lowered to 60 degrees and a trim speed of 74 KIAS was held during the turn to final. The angle of approach was easily controlled by varying the rate at which the wing was raised and the amount of power applied. The rate of descent was arrested immediately by power application. The power response provided good waveoff characteristics.

RECONVERSION AND VERTICAL LANDING

Reconversion and vertical landing were easily accomplished (CR 2). The 10/60 degrees CA was normally maintained until approximately one-quarter mile from the touchdown point where the wing angle was increased incrementally to approximately 45 degrees. At approximately 200 feet from the desired touchdown point, the wing was raised at constant rate to the proper wing angle for hover. A minimum height of 20 feet was used for the reconversion to avoid the instabilities which occurred at higher wing angles in ground effect. The instabilities are discussed in the Short Landing section.

Deceleration was excellent with the wing acting as a speedbrake. A slight floating or ballooning tendency was present at the lower wing angles if the wing angle was increased too rapidly, but there were negligible trim changes throughout the entire reconversion. Power management in the approach required only minor adjustments once the desired rate or angle of descent was established. There was no abrupt change in power required during the reconversion. Some excess power over hover requirements will be necessary to allow for gusts and control deflection requirements.

During vertical landings, the aircraft descended through recirculation turbulence between 10 to 4 feet. This increased pilot activity slightly but caused no control difficulty. The descent was arrested with power application, and the aircraft settled gently to the ground. No ground resonance tendency was encountered. A reconversion time history is shown in figure 29, appendix I.

SHORT LANDINGS

The short landings were made with the tail propeller engaged, and the power was controlled by the collective lever. The wing angle desired for landing was set on a short final approach due to the lengthy time involved if the entire pattern was flown at high wing angles. Due to time limitations the tests were concentrated on the 35/60 degree CA. This CA offered the best STOL performance without encountering the instabilities which were present at higher wing angles. Gross weights ranged from 34 340 to 41 100 pounds.

The landing technique used was to fly the final approach with the fuselage level and accept the resultant trim speed. The desired sink rate was controlled by power. Directional control deteriorated in ground effect with increased wing incidence, but was satisfactory up to 35 degrees. At wing angles of 30 to 35 degrees there was a slight negative ground effect below about 10 feet. This was easily overcome by a small power application as the sink rate began to increase just before touchdown. Negative ground effect occurred in such close proximity to the ground that no hard touchdowns were experienced. The pilot's natural tendency to flare the aircraft had to be overcome as this effectively increased the wing angle above 35 degrees. This resulted in an increased rate of descent and a deterioration of lateral-directional stability. A fuselage level attitude was maintained to a solid touchdown where the long stroke main landing gear struts provided firm but soft touchdowns from sink rates which averaged 560 fpm (9.3 fps).

Upon touchdown, the collective lever was lowered to the bottom gate, unhooked from the throttles, and lowered rapidly to the bottom stop in order to set minimum pitch on the propellers. The time required to lower the collective lever ranged from 1 to 4 seconds with the average being 2.2 seconds. Although the XC-142A propellers had no reverse pitch, the minimum pitch provided effective deceleration at speeds above approximately 35 KIAS. The propeller minimum pitch was much more effective than the brakes as a deceleration device at higher speeds. The brakes were inadequate and faded badly during maximum performance landings. The braking should be improved. The brakes are also discussed in the Taxiing and Hydraulic System sections. (C 22)

Upon completion of the landing roll, the wing was normally lowered to 10 degrees and the tail propeller disengaged and braked for taxi.

Landing data were recorded with Fairchild Flight Analyzers. Wind data were collected and wind corrections were applied to determine true airspeeds. No other corrections were made. The landing performance is summarized in table VII, and presented in figures 11 and 12, appendix I. A 35/60 degree CA landing time history is shown in figure 22, appendix I.

TABLE VII

LANDING PERFORMANCE COMPARISON

35/60 Degrees Conversion Angle

Weight (lb)	Ground Roll (ft)		Touchdown Speed (KTAS)		Total Distance over 50 feet (ft)		Speed at 50-ft Height (KTAS)	
	Flight Manual	Test	Flight Manual	Test	Flight Manual	Test	Flight Manual	Test
41 100	145	395	27	40	420	860	27	36
37 474	130	290	25	38	380	650	25	35
34 340	120	260	24	36	360	560	24	33

NOTE: See figures 11 and 12 of appendix I for test conditions.
A landing time history is shown in figure 22, appendix I.

The landing distances over a 50-foot obstacle were too long for an aircraft of this type and were not compaatible with the short takeoff distances. In all cases tested, the distances were greater than predicted in the Flight Manual. At 41 100 pounds, the ground roll was 170 percent (250 feet) higher than predicted. This increase was attributed primarily to the higher than predicted landing speeds, poor braking, and a lack of adequate negative thrust. The landing distances should be reduced to be compatible with the short takeoff distances. (C 21)

At the lightest weight tested of 34 340 pounds, the XC-142A did not meet the short landing criteria of 500 feet total distance over a 50-foot obstacle.

The highest CA used for landing without the tail propeller operating was 20/30 degrees. The only difference noticed by the pilot was that the stick hit the aft stop during the landing flare. A time history of this landing is presented in figure 23, appendix I.

A military pilot accompanied the contractor pilot on tests to determine XC-142A takeoff and landing characteristics at wing angles greater than 35 degrees. The test gross weight was about 38 200 pounds and the center of gravity was at 21-percent MGC. The SAS was on for these tests. The takeoff characteristics were previously discussed in the Short Takeoffs section. The landings for these tests were made at CA's of 40/60 and 45/60 degrees.

The first landing was made at a CA of 40/60 degrees. It was normal down to a height of about 15 feet where directional stability and controllability deteriorated noticeably (CR 4). Between 5 and 10 feet above the ground a very strong negative ground effect was experienced which was easily counteracted by adding power.

The second and final landing was made at a CA of 45/60 degrees and resulted in substantial damage to both outboard ailerons and both outboard engine exhaust nozzles. A power approach was established at about 200 feet height above the ground. The sink rate was stabilized at a torque setting of about 70 percent. Torque had been increased to 75 percent at 25 feet. At this height the aircraft began to react as on the previous landing, but stability and controllability deteriorated at an alarming rate until at about 10 feet it was apparent that a hard landing was inevitable. No attempt to go around was made, and both the pilots doubted it could have been made if attempted. Ground contact was made in a left rolling sideslip on the left main landing gear at a sink rate of about 12 fps. The aircraft continued its left roll until the outboard aileron and the No. 1 engine nozzle contacted the runway. The nose gear then touched down, and the aircraft rebounded into a right roll. The right main landing gear touched down and the left main landing gear lifted off the ground again. The right outboard aileron and the No. 4 engine nozzle then contacted the runway. The aircraft then settled down and was stopped without further incident. The initial impact was hard and could easily have resulted in other structural damage to the landing gear or wing attachment points. Control activity increased throughout the last 15 feet of the descent, and full lateral and directional control inputs were inadequate to keep the wings from contacting the runway. Records indicated roll attitudes were about 20 degrees and heading excursions were about 5 degrees (CR 8). The STOL CA limits should be determined. A time history of this landing is presented in figure 24, appendix I. (C 23)

The negative ground effect and stability and control deterioration experienced above wing angles of 35 degrees was thought to be caused by flow splitting. Flow splitting is defined as that point where the deflected propeller slipstream begins to precede the aircraft. These handling qualities were unsatisfactory and limited the aircraft performance capabilities to those attainable with 35 degrees of wing. After these tests, the contractor investigated the phenomenon from high wing angles, starting at 90 degrees. Instabilities began to occur at a wing angle of 80 degrees in ground effect. This 40- to 80-degree wing angle region may be a tilt-wing concept limit and should be further investigated. A possible solution may be a revised flap program. This phenomenon did not affect conversion or reconversion above 25 feet, out of ground effect.

STABILITY AND CONTROL TESTS

CONTROL SYSTEM FRICTION AND BREAKOUT FORCES

Control system friction and breakout force measurements were conducted in a closed hangar using an auxiliary power unit to energize the aircraft electrical system and both hydraulic systems. An inclinometer was placed on the wing and UHT to cross-check longitudinal trim and wing angle indicators. Breakout forces were measured at various wing angles with neutral, 1/2-nosedown, and 1/2-noseup trim settings. The results of these tests are compared with various specification limits in table VIII.

At a wing angle of zero degrees all breakout forces were within the limits of MIL-F-8785(ASG) and were satisfactory. At wing angles of 40 to 60 degrees the longitudinal and lateral breakout forces were within the limits of AGARD 408 and were satisfactory. The directional breakout forces exceeded these requirements. At a wing angle of 90 degrees, the longitudinal and lateral breakout forces were within the limits of AGARD 408, while longi-

TABLE VIII

CONTROL SYSTEM FRICTION AND BREAKOUT FORCES

Control	Longitudinal Trim	Forces at Various Wing Angles (lb)				Pilot Comment	MIL-F-8785(ASG)	MIL-H-8501A	AGARD 408
		0 deg	40 deg	60 deg	90 deg				
Longitudinal	Neutral	2(1)*	1.0	..0	2.0	Accept.	0.5 to 5.0	0.5 to 1.5	0.5 to 2.5
	½ nosedown	1(4)*	0.5	0.5	0.5	Accept.			
	½ noseup	2(3)*	2.0	0.5	2.0	Accept.			
Lateral	Neutral	2	1.0	1.0	1.0	Accept.	0.5 to 4.0	0.5 to 1.5	0.5 to 2.0
Directional	Neutral	10	12	≈11	≈11	**	1.0 to 14.0	3.0 to 7.0	1.0 to 10.0

* Parentheses denote tail propeller in brake position. All other longitudinal values were with the tail propeller in the engaged position.

** The high force gradient masked the breakout forces from the pilot.

tudinal breakout forces exceeded the requirements of MIL-H-8501A at trim settings of neutral and one-half noseup but were acceptable. The directional breakout forces failed to meet the minimum requirements of both AGARD 408 and Military Specification MIL-H-8501A and should be decreased. Acceptability of the breakout forces was determined from pilot comments, regardless of specification requirements.

LONGITUDINAL TRIM CHANGES

Longitudinal trim changes due to wing and flap movements were qualitatively evaluated and found to result in less than 5 pounds stick force (CR 2). The trim changes associated with landing gear extension at constant altitude are listed in table IX.

TABLE IX
LONGITUDINAL TRIM CHANGES
FOR LANDING GEAR EXTENSION AT CONSTANT ALTITUDE

Trim Speed (KCAS)	Altitude (ft)	Flaps (deg)	Power	Wing Angle (deg)	Long. Feel Force Selected	Trim Change (lb)
174	1200	0	PLF	0	Hover	2 Pull
135	1300	30	PLF	0	Cruise	5 Pull

STATIC LONGITUDINAL STABILITY

Static longitudinal stability tests were conducted at a mid cg of 20.7-percent MGC in the cruise configuration from a trim speed of 205 KEAS at 10 710 feet. The test technique used was to stabilize at airspeeds above and below the trim airspeed at a constant power setting. The static longitudinal stability, as indicated by the variation of UHT position and longitudinal stick force, was positive but low. Test results are presented in figure 19, appendix I.

The trimmability and controllability were good (CR 2). Aircraft response to pitch and roll control inputs was good. Although the static longitudinal stability was low, it was easy to control airspeed during cruise (CR 3).

DYNAMIC LONGITUDNAL STABILITY

Aircraft longitudinal dynamic stability tests were conducted in the cruise configuration at a 20.7-percent MGC cg at 210 KCAS and 10 200 feet altitude. All SAS was phased out. The longitudinal short-period was deadbeat at these test conditions and was satisfactory.

Longitudinal long-period (phugoid) tests were conducted at the test conditions listed above by displacing the aircraft from trim by $\pm$ 10 knots. The resulting oscillations were lightly damped with a period of 47 seconds, which was satisfactory.

LONGITUDINAL FEEL AND TRIM SYSTEM

The XC-142A was equipped with an irreversible fully-powered control system. For cruise flight, the longitudinal stick force gradient and trim rate varied with UHT trim position. The gradient was increased and the trim rate was decreased with increasing nosedown trim (increasing speed). In the V/STOL flight regime with the tail propeller switch in the "beta" position, the longitudinal feel and trim linkages were positioned at a minimum gradient. Trim rate was constant and the force gradient was essentially linear with both rate and gradient independent of UHT trim position and wing angle.

The cruise feel force gradients were increased during this evaluation to give better high-speed handling qualities. This resulted in objectionable control forces during landing with the tail propeller braked. To alleviate

this problem, a feel force switch was installed in the cockpit to enable selection of hover feel forces with the tail propeller braked. This change was satisfactory. Further discussion is included in the Maneuvering Flight section.

MANEUVERING FLIGHT

The maneuvering flight characteristics of the XC-142A were investigated at a cg of 20.7-percent MGC and at one gross weight during steady turns in the cruise configuration at 210 KCAS and 11 000 feet. The longitudinal stick force gradient (14.5 pounds per g) was below the minimum requirements of MIL-F-8785(ASG) for a class II (cargo) airplane. It was within the limits of LTV Estimated Flying Qualities Report No. 2-53310/4R939 which specified the limits for a class III (fighter) airplane. Although the longitudinal stick force gradient met the design requirement, the design limit load factor of 3 g's could be attained with about 29 pounds pull and about 1 inch movement of the stick. If the aircraft was not kept in trim during level flight acceleration, the stick force per g would be considerably lighter. This was a fighter type force gradient in an aircraft designed to a cargo type load factor and was unsatisfactory (CR 4). The gradient should be increased to meet the minimum requirements of a class II airplane in the cruise configuration. Test results are presented in figure 20, appendix I. (B 17)

The flight envelope limits existing during this evaluation (245 KEAS and 2 g's) were too restrictive to adequately define the XC-142A maneuvering flight characteristics. These limits should be expanded to at least 300 KEAS and 2.4 g's for Category II tests, but even more desirable would be V_{max} at 3 g's. (C 4, D 13)

DIRECTIONAL STABILITY IN CRUISE

Static directional stability characteristics were investigated in steady sideslips at a trim airspeed of 208 KCAS at an altitude of 10 210 feet. The sideslip angle limit established by the contractor was ± 5 degrees. The maximum cruise sideslip limits should be determined. (C 7)

The rudder deflection, force required, and the static directional stability were low (CR 4). There was a tendency to overcontrol, since the directional control effectiveness was high. The rudder forces should be increased for cruise. (C 18)

The dihedral effect and side force characteristics were positive and essentially linear with changes in sideslip angles up to ± 5 degrees. The side forces at 5 degrees of sideslip were strong enough that the pilot would not fly there without some reason. The static directional stability was slightly positive over the range tested. It could be considered linear within the range of normal data scatter, although a flattening of the curve (rudder position versus sideslip angle) can be observed in figure 18, appendix I. The longitudinal stick force change caused by sideslip was satisfactory up to sideslip angles of 5 degrees.

Dynamic lateral-directional stability was evaluated by making releases from 5-degree sideslip angles. The resulting oscillations damped completely in three-quarters of a cycle, but the time to damp was 7.5 seconds which was objectionably long to the pilot (CR 5).

LATERAL CONTROL

Qualitatively, the lateral control effectiveness was quite high, especially for a transport

type aircraft (CR 2). The maximum safe roll rate limits should be determined. (C 8)

STALL APPROACHES

Full stalls were not made in the XC-142A, but power-on stall approaches to heavy buffet were made at the configurations shown in table X. Stalls should be investigated in the cruise configuration and in a 10/30 degree CA landing configuration. (C 6)

TABLE X

STALL APPROACH SUMMARY

Tail Prop Braked Landing Gear Up

cg = 21-percent MGC 10 000 feet

Power for Level Flight

Conversion Angle (deg)	Gross Weight (lb)	Prop (pct rpm)	Engine shp	Trim Speed (KIAS)	Buffet Onset Speed (KIAS)	Minimum Speed Attained (KIAS)
0/0	36 500	75	760	155	147	122
0/30	36 300	91	990	107	90	68
14/30	36 000	91	920	77	64	58

In the cruise configuration, buffet onset occurred at 147 KIAS and increased to heavy buffet at 122 KIAS. Stability and controllability were good throughout. There was a slight deterioration in roll control at 122 KIAS. The stall approach was terminated at this speed because of the buffet intensity. A time history of this stall approach is presented in figure 25, appendix I.

At a CA of 0/30 degrees, buffet onset occurred at 90 KIAS. There was light buffet down to 68 KIAS where the aircraft pitch attitude was approximately 12 degrees nose-up. The stall approach was terminated at this speed because the No. 2 engine oil was overheating.

At a CA of 14/30 degrees, buffet onset occurred at 64 KIAS. The stall approach was terminated at 58 KIAS in heavy buffet at a pitch attitude of 11 degrees nose-up. Control about all axes was still good and recovery was immediate upon application of power. A time history of this stall approach is presented in figure 26, appendix I.

For the configurations investigated, recovery from prestall buffet was immediate upon power application. The prestall handling characteristics and recovery from prestall buffet were good. Stall warning characteristics were good, but in the cruise configuration the airspeed margin for stall warning (25 KIAS) was too large and the buffet onset speed (147 KIAS) was too high. Because of the high buffet onset airspeed, light buffet was encountered at the speed for maximum rate of climb. The buffet onset speed should be reduced in the cruise configuration. (C 20)

FLIGHT TEST INSTRUMENTATION

Existing contractor-installed Category I flight test instrumentation was used to record data during this evaluation. Although it was satisfactory for Category I aircraft development requirements, it will not be adequate for Category II tests to define specific aircraft characteristics. Additional parameters and improved accuracies will be required. The instrumentation should be modified to comply with Category II requirements. (B 78)

AIRSPEED CALIBRATION

A calibration of the XC-142A noseboom airspeed system was made in the cruise configuration over a range of airspeeds at 10 000 and 25 000 feet by the stabilized pacer method. The XC-142A indicated airspeeds and altitudes were obtained from AFFTC instruments which had been carefully selected for their low hysteresis and high repeatability. These instruments had been recently calibrated and were mounted in the XC-142A photopanel for these tests. The pacer was a calibrated AFFTC T-37.

The airspeed calibration of the XC-142A noseboom system is presented in figure 13, appendix I.

EMERGENCY OPERATIONS

GEARBOX OR PROPELLER FAILURE

The inboard propellers could be feathered separately, but both outboard propellers had to be feathered simultaneously. Any engine could be shut down without stopping the propellers. These operations were demonstrated in flight by the contractor. The contractor pilot's comments regarding aircraft handling qualities while feathering the two outboard propellers were as follows:

"At 5000 feet, 164 KIAS, landing gear was extended and the CSD generator was turned off to insure that the APU generator could carry the load. After resetting the yaw and roll stabilization systems and connecting the copilot's collective to the beta linkage, the No. 1 declutch and feather buttons were pressed. The No. 1 and 4 propellers were immediately observed going to a feathered position, and the No. 1 and 4 engines were shut down. Power was advanced on the No. 2 and 3 engines to maintain 160 KIAS while directional damping and roll control were briefly evaluated. No undesirable characteristics were observed but considerable "crosstalk" (unsynochronized propeller beat - Editor) was apparent between No. 2 and 3 propeller speeds. Flaps were extended to 30 degrees, and at 120 KIAS the emergency hover longitudinal feel forces were selected. With the copilot controlling propeller speed at 90 percent with the collective stick, lateral-directional characteristics were evaluated at wing-flap settings of 5/30 and 10/30. The descent boundary was investigated in both configurations. The 10/30 setting was better in roll control and yielded a 1000-fpm descent rate, buffet free, at 92 KIAS trim speed. A rather long final approach was flown at 10/30 with an 85-KIAS trim speed. Touchdown was made at 80 KIAS with the copilot applying flat propeller pitch as throttles were retarded to idle. With a propeller feathered and the respective gearcase stopped, the gearcase caution light glowed. Because there was only one caution light for the six gearcases, a subsequent high temperature or low pressure in an operating gearcase could go unobserved."

The contractor pilot's comments on aircraft handling qualities with one inboard propeller feathered were as follows:

"Following a 35/30 takeoff, the airplane was cleaned up, climbed to 5000 feet and stabilized at 164 KIAS with 75-percent propeller rpm. The landing gear was extended. The No. 3 declutch and feather buttons were pressed, and the No. 3 engine secured. Trim change was easily controlled with moderate rudder and aileron inputs. The directional damping and left and right 1/4-aileron rolls were evaluated. The collective was "hooked up" and the airplane slowed to 120 KIAS with 30-degree flaps. With power for level flight, full left rudder trim was required plus moderate left pedal input. Aileron trim input was moderate. Rudder required in this configuration increased to nearly full deflection as power was applied to climb. The wing was then raised to 5 degrees which alleviated a portion of the directional control problem except at climb power when near maximum rudder input was necessary. Again lateral control was adequate. With 10 degrees of wing and 30 degrees of flaps an immediate decrease in rudder required to trim was apparent, probably due to differential beta input. Roll control remained adequate and at a trim speed of 82 KIAS the buffet onset occurred at 600 fpm. From 600 fpm descent rate, a moderate power input was made resulting in 500 fpm climb with no adverse control requirements. The airplane was then returned to the field and landed in the 10/30 configuration. At the touchdown speed of 80 KIAS, moderate right rudder and aileron were required to counteract the nose left swerve tendency."

GOVERNOR FAILURE

Failure of the propeller governor was demonstrated as part of the military crew checkout. This failure was handled in a routine manner by employing "beta backup" procedures where the copilot, through collective lever movement, controlled propeller rpm as power was being changed by the pilot. This procedure was satisfactory and it appeared that most missions could be completed in this manner.

ENGINE FAILURE

There were no changes in aircraft handling characteristics following engine failure or shutdown in cruise. No engines were shut down in hover. The cross-shafting should prevent any control difficulties in the event of an engine failure during hover.

SAS OFF OPERATION

Dual rate damping augmentation was provided about all three control axes, and dual attitude hold was provided abou the pitch and roll axes. Pitch rate stabilization was provided when the tail propeller was engaged. Roll rate and attitude, yaw rate, and pitch attitude stabilization were provided when the wing was above zero degrees; stabilization authority was zero at zero degrees of wing, and was phased in linearly with wing angle becoming 100 percent effective at 15 degrees and above. A single channel altitude rate damper was provided. Flight with single channel and dual channel failure about each axis and all channels disabled was demonstrated by the contractor over the entire operating regime from hover through conversion and reconversion. Generally speaking the aircraft control was satisfactory for the test conditions; however, the degree of pilot skill required to satisfactorily control the aircraft with SAS off increased sharply with gusting wind conditions. Caution was necessary to avoid developing appreciable rates, especially about the roll axis.

Although no complete vertical takeoffs and conversions were made with all SAS off by military pilots, several were made with one-half SAS in all axes, and on one conversion, complete SAS was lost and reengaged several times. There was very little deterioration in aircraft flying qualities with one-half SAS except for a slight but noticeable decrease in roll damping. The 1/2-SAS condition did not require any unusual pilot inputs and the aircraft was easily controlled. The contractor later demonstrated a complete vertical takeoff, conversion, reconversion, and vertical landing (verticircuit) successfully with all SAS off.

Time histories of a SAS OFF conversion, a reconversion, and a hover are shown in figures 28, 30, and 31, respectively, in appendix I. The allowable SAS off flight limits should be completely defined. (C 10)

The contractor pilot's comments and conclusions on SAS OFF hovering and converisons were as follows:

"Handling characteristics while in the hover were satisfactory with 1/2-axis operation in pitch and roll and with the yaw axis inoperative. Total pitch system out, in combination with one-half roll and all other axes out required extra pilot attention to pitch attitude control and some minor excursions from the desired attitude were considered normal. Roll axis out or complete SAS off characteristics required total pilot attention in attitude control with the emphasis on roll attitude control. Strong roll inertia was present which had to be anticipated when making bank angle corrections. Pilot learning in the all axes "off" configuration was evident and may influence the Cooper Rating as learning increased.

"Conversions and reconversions with the SAS inoperative were acceptable for emergency conditions. One-half SAS inoperative in any combination of axes presented no difficulty in control and only a small change in flight characteristics. Complete SAS off in yaw axis was reflected in a moderate increase in rudder pedal activity, particularly when near the ground in the hover. Complete SAS off in the pitch axis was characterized by a slight tendency toward PIO[4] due to high control effectiveness and light stick forces. Complete SAS off in the roll axis was the most critical due to the high control power available coupled with high inertia effects. The natural period in roll was relatively long and allowed the pilot to act as the stabilizing influence. If the pilot generated significant roll rates intentionally, correction to a zero bank angle required considerable effort. The aircraft had to be flown smoothly and bank angle corrections performed at reduced rates to minimize roll inertia effects."

SYSTEMS EVALUATION

■ AIRFRAME

● DESCRIPTION

The fuselage was of semi-monocoque construction and was divided into the nose, mid, and aft sections. The cockpit contained a side by side pilot and copilot seating arrangement with zero-zero rocket-type ejection seats.

The wing was mounted high on the fuselage and was capable of being tilted upward to a maximum angle of 98 degrees from the horizontal for hover and transition. The wing group consisted of the

[4] Pilot induced oscillation.

basic wing structure, ailerons, leading edge slats, trailing edge flaps, engine-propeller cross-shafting, gearboxes, and four engine nacelles.

The tail group included a unit horizontal tail (UHT), vertical stabilizer, rudder, and tail propeller boom. The tail propeller boom consisted of a structural member which was attached to the aft fuselage section below the vertical stabilizer and extended aft of the rudder. The tail propeller shafting was housed in the boom, and the tail propeller gearbox and propeller were attached to the end of the boom. The tail propeller was oriented in a horizontal plane to provide pitch control in VTOL, STOL, and hover.

The cargo compartment was 7 1/2 feet wide, 7 feet high, and 30 feet long and was designed to accommodate 32 fully-equipped troops or 30 fully-equipped paratroops. Access to the cargo compartment was through the aft cargo door. A loading ramp and two detachable dock boards were used for loading operations. Floor-mounted cargo tiedown fittings with 5000- and 10 000-pound capacities were located at regular intervals.

The aerial delivery equipment proposed for the aircraft consisted of roller conveyors that could be installed on each side of the cargo compartment floor and ramp, and a pendulum extraction system to discharge an extraction chute from the aircraft. This equipment was not available for evaluation.

FUNCTIONAL ANALYSIS

escape hatch

The rear escape hatch tore loose during flight at approximately 280 KIAS and caused some damage to the leading edge of the UHT. Loss of the door was attributed to a structurally weak hinge point which should be strengthened on all XC-142A aircraft to withstand airspeeds through V_{max}. (C 30)

windshield

Above 215 KIAS, the windshield vibrated at a low frequency which could be felt with the hand and was visible on the surface of the windshield. This vibration caused a pulsating pressure in the cockpit. The strengths of the windshields should be reevaluated, and the effects of the vibrations on their fatigue life should be determined. The cause of the vibrations should be determined and the vibrations should be eliminated. (C 24)

The windshields of the XC-142A aircraft were not impact-proof. Any production aircraft should be equipped with impact-proof windshields. (D 25)

The large amount of window area in the cockpit caused an extreme temperature rise inside the cockpit. On a warm clear day the temperature rose as much as 35 degrees F above ambient. This temperature differential was reduced by painting the overhead window panels. This paint eliminated overhead visibility. Windows of this type should be installed on V/STOL aircraft to permit overhead visibility. Consideration should be given to using curtains on these windows, but they should be compatible with ejection seat operations. (D 34)

walkways

The walkways on top of the fuselage and wing appeared to be adequate. The upper cargo door (in the up position) was also utilized as a walkway by the maintenance personnel. A walkway should be provided on this surface

if it would not interfere with the airflow. If a walkway is not feasible, a protective cover should be provided. (C 32)

personnel entrance door

The personnel entrance door opened inadvertently in flight. This was attributed to the lack of an overcentering lock. The door lock was modified to include an overcenter mechanism. The overcenter lock should be incorporated on all XC-142A aircraft. (B 28)

access panels

Many access panels were attached with screws rather than Dzus or camlock fasteners. One example was the access panel to the connectors for the hydraulic power cart which required frequent removal, yet contained 96 screws. Unless access panels are stress panels, they should be attached with Dzus or camlock type fasteners to facilitate maintenance. (D 33)

fuselage shaft tunnel

Failure of the tail propeller shaft could damage the fuselage shaft tunnel in the fuel cell area and rupture the fuel cells. Shielding should be provided in the fuel cell area on production aircraft. (D 31)

front air deflectors

On several occasions the mechanically-actuated front air deflector operated out of sequence and was damaged when the wing was lowered. The deflector was designed to follow wing movements through the use of bellcranks and pushrods. A reliable system of actuating the front air deflector in the proper sequence should be incorporated. (B 29)

The area near the wing and air deflectors on top of the fuselage was dangerous to maintenance personnel due to movement of the air deflectors and wing.

wing

There were two failures during the static wing structural tests at approximately 110 percent of the limit load for a 50-fps gust at 350 KEAS. A portion of the upper skin buckled and severely damaged ribs, stringers, and spars. The static wing should be modified and tested to the desired 150-percent limit load. (D 27)

Wing ribs on the aircraft cracked during the 50-hour tiedown test and during the early portion of the flying program. These cracks were attributed to vibration. Redesigned ribs were installed on the No. 1 aircraft wing and the static wing. The wings on the remaining aircraft should also be modified. (C 26)

unit horizontal tail (UHT)

The UHT torque box skin cracked along several lines of rivet holes. Some cracks were as long as 6 inches. The reason for failure was a combination of high vibration and inadequate riveting techniques. The general quality of the skin installation was poor and the skin buckled between rows of rivets. All the UHT's were modified by strengthening the torque box skin. The vibration level was reduced by improving the airflow over the UHT through the addition of nacelle oil cooler cutout doors.

nacelle cooling

During the first few flights of the XC-142A, high temperatures were indicated by the engine fire warning lights in all four nacelles. Investigations revealed a reverse flow of exhaust gas into the nacelle area due to low pressures in the

nacelle. The problem was corrected by extending the engine tailpipes and providing additional nacelle seals.

engine oil temperature

During early flights of the XC-142A, high engine oil temperatures were encountered during operation with wing incidence angles greater than 20 degrees. The wing had to be lowered to allow the temperatures to decrease. The problem was relieved by cutting cooling air exits just behind the oil cooler. Hydraulically-actuated doors were installed to close the ducts for cruise.

cargo loading

A winch was not provided to facilitate cargo loading. A production cargo aircraft should have a winch. It should be determined if the existing cargo tiedowns are satisfactory as attachment points for snatch pulleys to facilitate loading cargo into the cargo compartment. A snatch pulley arrangement would allow routing of the cable from the cargo through the front personnel entrance door. The cargo could then be pulled into the cargo compartment by a vehicle. If present tiedowns are unsuitable, hard points should be provided for snatch pulleys. (D 37, D 38)

aerospace ground equipment (AGE)

There was no opportunity to evaluate the adequacy of jack pads, dollies, hoists, etc. It was noted, however, that wing jacks were not required. The aircraft could be jacked with fuselage or axle jacks. The jacks used by the contractor were not in the Air Force inventory since they had to be short and yet have a long stroke. The required jacks should be provided. (B 35)

PROPULSION

DESCRIPTION

The XC-142A was powered by four T64-GE-1 turboshaft engines, each rated at 2850 maximum shaft horsepower. Each engine drove its corresponding integral gearcase (IGC) and propeller through an overrunning clutch. Cross-shafting driven from the IGC allowed all four propellers to be driven by a single engine. Each engine was started by using hydraulic power from another engine or from the APU starter-pump.

A retractable air inlet screen was mounted in each nacelle to prevent entry of foreign objects. These screens were in place with the landing gear down, and retracted with the landing gear up.

FUNCTIONAL ANALYSIS

engine stall characteristics

Rapid throttle transients during flight did not induce compressor stalls in the T64-GE-1 engine under the conditions investigated. Table XI shows a summary of rapid throttle transient tests accomplished by the contractor.

TABLE XI

RAPID THROTTLE TRANSIENTS

Throttle Movement	Time (sec)	Airspeed (KIAS)	Altitude (ft)
1/4 - Full	2	215-225	10 200
Full - 1/4	2	225-215	10 200
1/4 - Full	2	215 230	10 100
Full - 1/4	2	230-215	10 100
1/4 - Full	2	165-180	10 100
Full - 1/4	2	180-165	10 100
1/4 - Full	1	- -165	10 100
Full - 1/4	1	- -165	10 100
1/4 - Full	3	165-180	10 000
Full - 1/4	3	180-165	10 000
1/4 - Full	4	165-180	10 000
Full - 1/4	4	180-165	10 000

Compressor bleed-valve operation was controlled by the engine starting circuit and the bleed valve could not be manually operated. Thus, compressor bleed valve effect on rapid throttle transients could not be determined. Compressor stalls occurred only during ground start cycles and as a result of compressor bleed-valve malfunction.

free turbine lubrication

During engine shutdown in flight the drag induced by the starter-pump was sufficient to prevent or limit windmilling of the gas generator turbine while the power turbine was free to rotate. Table XII shows a summary of in-flight engine data from the No. 3 aircraft. From these data it can be seen that even though there was no rotation of the gas generator turbine, the overriding clutch between the IGC and the power turbine had sufficient drag to cause rotation of the power turbine.

TABLE XII

XC-142A WINDMILLING ENGINE DATA

N_f rpm	N_g rpm	Airspeed (KIAS)	Altitude (ft)
1341	529	220	15 400
1343	0	188	15 200
1352	16	164	18 800
1109	20	182	15 200
1123	9	171	15 200
1769	46	186	25 500
1361	0	178	25 600

Since the power turbine did not stop rotating and its lubrication was dependent upon the oil pump driven by the gas generator turbine, the adequacy of the power turbine bearing lubrication was questionable. Additional studies should be made to determine the adequacy of power turbine lubrication when the gas generator turbine is not rotating. Operating restrictions should be established to prevent engine damage. This was a serious problem since recommended cruise operation was with two engines shut down. (C 60)

starting time

Initial engine starts were performed using the APU hydraulic pump assist to an inboard engine. Available manuals indicated that starting times should not exceed 35 seconds (to idle rpm) for a standard day. Initial starting times were almost double these indicated maximums. Either the pressure losses were too great or the starter pump capacity was too low to meet the desired start times. The starting times should be reduced to 35 seconds or less. (D 57)

Cold start data (below 20 degrees F) were needed to determine if start times would be appreciably extended at lower temperatures and if any adverse engine start parameters were encountered as a result of the longer starting times.

engine topping

The procedure for topping the engines was to set the engines for maximum turbine inlet temperature (T_{t5}) with the collective lever at the upper gate. This procedure gave the pilot no capability for exceeding takeoff power in an emergency. The engines should be rigged so that takeoff power can be exceeded by pulling through the upper collective gate. This would add to the safety margin in case of an engine failure or other emergency conditions. (C 103)

airstarts

Airstarts could not be accomplished from a windmill condition because the gas generator turbine did not windmill fast enough. All airstarts were accomplished by using hydraulic crossfeed pressure

from an adjacent engine. Table XIII shows a summary of airstarts completed by the contractor. All starts were successful and within acceptable time limits as stated in the engine operating instructions.

of the compressor inlet and/or the possibility of incorporating a separator should be considered as aids in reducing the FOD rate. (C 56)

TABLE XIII

XC-142A USAF S/N 62-5923 T64-GE-1 AIRSTARTS

Engine Started	Source (engine)	Altitude (feet)	Airspeed (KIAS)	Time to Idle (sec)*	Cold Soak Time (min)
4	3	15 000	216	35	1
4	3	15 000	182	38	10
4	3	20 000	199	35	1
1	2	15 000	162	35	36
4	3	10 000	216	40	20
4	3	10 000	217	33	-
1	2	2 100	225	-	-
1	2	6 000	150	-	-
1	2	5 500	207	-	-
4	3	5 500	207	-	-

*Time to idle is the time from beginning of N_g to idle N_g.

foreign object damage (FOD)

As of 1 March 1965 there was a total of 1576 hours of engine operation. Of this total, 1543 hours occurred during aircraft ground operations and 33 hours during flight. Ten engines had received FOD by this time. Three of these engines were damaged at LTV, two in ground tests, and one in hover flight. Six engines were damaged during testing at Hamilton Standard Division (HSD) and one at Hiller Aircraft. Not all of the FOD was evident prior to engine teardown. The primary causes of FOD were small metal objects such as nuts, bolts, and rivets. Airflow patterns using various wing angles, power settings, etc., should be studied in an effort to define those configurations that are likely to cause FOD. Redesign

The T64-GE-1 turboshaft engine was susceptible to ingestion of oil or hydraulic fluid. The IGC and propeller hub were located so that fluid leaking from these components was drawn into the compressor and caused contamination. Noticeable power losses resulted from the ingestion of this hydraulic fluid. LTV incorporated a cleaning process using Rust-Lick 606 (FSN 6850-066-2333) to clean the engines. This process satisfactorily restored engine performance, although elimination of the hydraulic and oil leaks from the propeller hub and IGC would be a better solution. If engine cleaning with Rust-Lick becomes a necessity during Category II, a simple means should be used for determining the allowable power loss and corresponding interval at which the engine should be cleaned. The procedure used by

LTV required a ground runup of each engine with the wing in a horizontal position. Ambient temperature, torque, and pressure altitude were recorded for a set propeller rpm. The corrected shaft horsepower was then computed and compared with the engines' reference curves to determine the power loss. The transfer valve and IGC hydraulic leaks should be corrected. (C 61)

spectrometric oil analysis

LTV performed a spectrometric oil analysis on the engines and gearcases after every 5 flight hours. The oil analysis served to detect buildup of metals from worn or damaged components in order to predict impending failure. Results of the LTV analysis were not complete.

A spectrometric oil analysis program should be established during the Category II test program to predict and therefore prevent gearcase failures. Oil samples should be taken from the gearcases and engines after each flight. Oil analysis data collected at LTV should be used as a guide in establishing threshold limits and trends.

was applied to the cross-shafting at the IGC's so that if one or more engines were shut down, the remaining engines would supply power to all four main propellers and the tail propeller.

Shafting from each wing was interconnected by the tridirectional gearcase located in the center wing section. This gearcase transmitted power to the pivot gearcase. The pivot gearcase transmitted power to the tail propeller and accessory drives.

Decouplers were used to disengage No. 2 and 3 main propellers from the power transmission system. The propellers could then be driven directly by the No. 2 and 3 engines. Number 1 and 4 main propellers could not be disengaged from the transmission system. A tail propeller clutch engaged and disengaged the tail propeller. A brake system was used to stop the tail propeller.

Each of the four main propellers had four variable pitch blades. A master governor control maintained propeller speed. The blades consisted of fiberglass airfoils bonded to steel spars. The inboard main propellers could be feathered individually after being decoupled. The outboard propellers had to be feathered together after the inboard propellers were decoupled from the cross-shafting.

The tail propeller was used to control pitch attitude during all V/STOL flight modes. The propeller consisted of three variable pitch blades and a control hub. The blades had symmetrical airfoils to provide either positive or negative thrust and were constructed similarly to the main propellers.

front frame cracks

Three T64-GE-1 engines were removed for front frame cracks. All three cracked front frames occurred during testing at the Propulsion Integrated Test Stand (PITS). The General Electric Company determined that the cracks resulted from fatigue, and was developing a "beefup" to prevent further front frame cracking. Exact cause of the cracking or extent of the "beefup" was not known at the time of the NPE. The front frame should be modified to prevent cracking. (B 55)

engine and IGC oil tanks

A persistent problem was breaking of the straps which held the engine and IGC oil tanks. It was also difficult to position the tanks during installation while using the straps. Installation of the engine and IGC oil tanks should be simplified. (B 59)

The oil tanks cracked and leaked early in the program due to the lack of load pads on the tanks. All engine and IGC oil tanks should have load pads to prevent cracking. (B 58)

■ POWER TRANSMISSION AND PROPELLERS

● DESCRIPTION

The power transmission system interconnected the four main propeller IGC's and the tail propeller gearcase. The system consisted of cross-shafting, a tridirectional gearcase, and a pivot gearcase, and transmitted power from the engines to the propellers and aircraft accessories.

The cross-shafts were located in the leading edge of the wing between the engines and in the top center of the fuselage from the wing to the tail propeller. The shafts were connected to the gearcases by flexible ball spline couplings which compensated for flexing of the airframe. Power

● FUNCTIONAL ANALYSIS

operation

Number 1 and 4 engine shutdown and propeller feathering was demonstrated at 5000 feet and 164 KIAS. All systems functioned properly and safe landings were demonstrated. The feathering circuit was designed so that the propellers from engines No. 1 and 4 were feathered together to prevent an asymmetric flight condition.

The transmission system capabilities were also demonstrated by decoupling and feathering the No. 3 propeller and shutting down the No. 3 engine. All systems functioned satisfactorily.

tridirectional gearcase

The tridirectional gearcase was limited to a 10-hour tail propeller operation inspection interval. This restriction was imposed because of a fatigue failure of the input gear (P/N 210-75412). The fatigue was caused by a resonant frequency associated with main propeller rpm. A modified gear should be incorporated in the tridirectional gearcase to eliminate the fatigue failure problem and permit an increased inspection interval. (B 63)

integral gearcase (IGC)

The main IGC's were limited to a 25-hour inspection interval. This was due to the worn condition of the titanium liner in the front of the IGC. This liner was changed to steel to eliminate the wear problem. Pending results of the 25-hour gearcase inspection at LTV on 3 and 5 April 1965, the gearcase inspection interval was to be adjusted.

cross-shafting

The overall operation of the cross-shafting system was satisfactory. Most major discrepancies had been corrected. Since the reliability of the cross-shafting system was unknown, it was important that this system be inspected at frequent intervals. Inspection and maintenance were difficult because many of the inboard wing and aft fuselage bearings were practically inaccessible. Past inspections did not reveal many discrepancies. Some potential problem areas, such as the rotation of the fuselage shaft bearing at station 450,

should be closely monitored. All shaft bearings should be accessible in production aircraft. A complete maintainability program should be pursued to insure that these and all other components in a production aircraft are accessible. (D 65)

Wing and fuselage shaft bearing temperatures could not be monitored in flight. An indicating system should be installed to warn the pilot of an overheating condition until the reliability of the shafting has been proven. (A 2)

Inspections have shown excessive wear to the wing shaft bearings caused by failure of the pins which hold the outer race of the bearings to the pillow blocks. After failure of the pins, the outer race could not move freely which caused fretting corrosion of the outer race. The wing shaft bearing components should be modified to improve bearing life. (C 64)

propellers

The hub moment produced by the main inboard propeller blades was higher than expected. The resulting increased stress produced cracks in one IGC. The hubs on the No. 2 and 3 airplanes were to be modified to compensate for the increased stress and the No. 5 airplane was scheduled to have new hubs. The exact nature of the hub modification was not known. All main propeller hubs should be modified to compensate for the high stresses. (B 67)

Several cracks developed on the main propeller leading edges. Although the cracks were not considered critical and were reparable at LTV, a longer operative time was needed to determine the overall reliability of the propellers.

Contractor test data revealed a propeller thrust deficiency. The reasons for the deficiency were not fully understood. The propeller efficiencies should be improved. (C 14)

aerospace ground equipment (AGE)

There was limited opportunity to evaluate the propeller and transmission system ground safety locks. These locks consisted of fabric boots and straps positioned between two blade tips of one propeller and two blade tips of the adjacent propeller to prevent rotation of the propellers and transmission in the wind. The tail propeller brake, which was also used to brake the entire transmission system at 5-percent propeller rpm or less, could not be used while the engines or APU were not operating since it required a hydraulic pressure source. The locks appeared to be effective while towing the aircraft in winds of approximately 15 knots.

engine and gearcase mounting

Installation of the IGC mounting struts in the wing was very critical. There was a 0.002-inch (minimum) to 0.007-inch (maximum) tolerance on the ball mounting of the struts. LTV engineering recommended the 0.002-inch tolerance and stated that once the IGC installation was satisfactory, the mounting strut adjustments should not be moved. Since the mounting system was designed to be completely stable with one strut removed, freedom of movement of one strut should be a good check for proper installation.

A number of dimensions and tolerances had to be adhered to in order to properly locate the IGC and engine and a template was devised to aid in this installation. In one case, an improperly installed

engine resulted in binding of the cross-shafting and flex coupling. Precise installation procedures and techniques, needed to insure that the IGC mounting struts were properly located, should be provided. (D 66)

FUEL SYSTEM

DESCRIPTION

The fuel storage tanks were arranged in two clusters, each composed of a main and a transfer tank. The total usable fuel capacity was 1400 gallons. The forward cluster was located in the top of the fuselage immediately forward of the wing leading edge and the aft cluster was in the top of the fuselage directly behind the wing trailing edge. The two clusters were normally separate but could be interconnected through an emergency transfer line. A crossfeed valve allowed interconnection of the fuel manifolds to provide a common manifold for all four engines.

FUNCTIONAL ANALYSIS

refueling

The single-point refueling system was similar to other Air Force systems and was not difficult to use. About 11 minutes were required to service the forward and rear clusters with 1400 gallons of fuel.

defueling

Defueling both clusters through the single-point system required about 30 minutes. The forward cluster emptied first because of the higher total head pressure which resulted from the normal 1-degree-noseup ground attitude. Only the transfer tanks could be defueled through the topside fillers. The main tanks did not have fillers, and check valves between the tanks prevented crossflow.

The rear cluster held approximately 200 gallons more than the forward cluster when the aircraft was fueled by the single-point system. This was attributed by the contractor to a slightly lower float valve installation in the rear cluster.

A defueling check valve had to be manually positioned inside the aircraft prior to defueling. Failure to open this valve could result in collapse of the hose from the defueling vehicle and could cause possible damage to the fuel system up to the defueling check valve. The defueling valve should be located near the single-point refueling controls. If the location of this valve cannot be changed, a placard should be placed near the single-point refueling controls to remind ground personnel to open the valve. (B 68)

There were 16 valves used to drain fuel and water from the fuel system. They were located near the lower fuselage line and were easily accessible even though the fuel cells were located in the top of the fuselage.

ELECTRICAL SYSTEM

DESCRIPTION

The primary electrical power generating system consisted of two self-cooled brushless 25-kva generators. One generator was driven by the CSD, powered from the pivot gearcase accessory drive, and the other generator was driven by the APU. Either generator was capable of furnishing the entire electrical load requirement in all modes of operation. Both generators were operated during takeoff, landing, hover, and transition.

Aircraft dc electrical power was provided by two 28-volt transformer rectifiers. Emergency electrical power was supplied by a

4-kva ac and a 28-volt dc generator driven by a constant speed variable displacement hydraulic motor. A rechargeable battery provided ignition power for the APU.

FUNCTIONAL ANALYSIS

power

The two main electrical power source circuits (CSD and APU generators) were frequency-locked but not phase-locked and the control circuit was designed so that the two sources could not be connected simultaneously to any one bus. Several problems, such as SAS cut-outs, were caused by transient currents. The electrical circuits in a production aircraft should be designed to prevent transient currents from affecting other systems. (D 72)

When hydraulic power was applied to PC 2, all essential buses were energized even when external electrical power was not connected and the CSD and APU generators were not operating. This was caused by operation of the emergency generator which was powered by PC 2.

tail propeller clutch control circuit

A major problem arose with the tail propeller clutch system. The original system included a time-delay which allowed 20 seconds of operation at 475-psi hydraulic pressure. This was necessary to allow the tail propeller to come up to speed before application of the full 3150-psi hydraulic pressure. Transient currents caused the time-delay relay to function prematurely and apply 3150 psi to the clutch which overtorqued and failed the tail propeller shafting. The system was modified to include a separate sequencing switch and time-delay which appeared to alleviate the problem. However, the wrong switching sequence was inadvertently used which also resulted in a shaft failure. A reliable and safe tail propeller clutch control system should be installed on all aircraft. A tail propeller on-speed indicator should be considered to preclude premature application of full hydraulic system pressure to the clutch. (C 69)

propeller monitor

A high abort rate was caused by malfunctioning of the seizure warning circuit of the propeller blade pitch control rod swivel. The malfunctions were caused by a high vibration environment which resulted in broken wires in the warning circuit and a false propeller seizure indication. This problem should be corrected prior to Category II testing. (B 73)

stability augmentation system (SAS)

The SAS was frequently inoperative for two primary reasons. Vibration caused breakage of wires leading to the SAS at the wire clip attachments. The SAS was also frequently inoperative because of transient currents. A failure in one channel would cause the monitor to shut off both channels and the pilot could then reengage the good channel with the collective lever switch. Wire breakages should be eliminated. The electrical circuits should be designed to eliminate transient currents. SAS-OFF characteristics are discussed in the Emergency Operations section. (B 73, D 72)

feathering circuit

The feathering circuit required that the decouple switch(es) for No. 2 and 3 engines be energized prior to activation of the feathering switches for No. 1 and 4 propellers. During an emergency the decoupling step was omitted and

feathering was not accomplished. The decoupling operation should be automatic and should be integrated with the appropriate feathering switches. The feathering circuit should also be designed so that it can be checked during preflight. (D 70, D 71)

gearcase oil pressure caution light

When a propeller was feathered, the integral gearcase oil pressure caution light illuminated and remained on as long as the propeller remained feathered. There was only one oil pressure caution light; therefore, monitoring of the remaining gearcases was not possible while a propeller was feathered. The wiring should be changed to deactivate the warning circuit of an inoperative gearcase. (D 76)

lighting

An evaluation of the internal and external lights was not made because night flying was prohibited. The external lights consisted of anticollision, navigation, and landing lights. A production aircraft should also include a controllable searchlight(s). (D 77)

battery

Prior to engine start, the APU had to be started or an external electrical power source used to provide electrical power. The battery in the XC-142A could be used only to supply ignition power to the APU. An investigation should be accomplished to determine the feasibility of connecting the battery to the electrical bus system. (D 74)

During APU operation, the generator switch had to be in the ON position unless electrical power was being supplied to the bus system by the CSD generator or external power. Failure to do so would result in complete discharge of the APU battery.

nose gear

Nosewheel steering was ineffective at the beginning of the flight test program because the followup transducer was or a separate power source from th input transducer. The nosewheel steering became satisfactory aft both transducers were [illegible] the same power source.

■ HYDRA[illegible]EM

● DESCRIPTION

The hydraulic system consisted of two separate power control systems (PC 1 and PC 2), a utility hydraulic system, an emergency utility hydraulic system, and an engine start system. Each of the systems operated at 3150 psi hydraulic pressure.

PC 1 was a completely independent system driven by two starter-pumps, one mounted on each right-hand power plant. The system provided hydraulic pressure to one-half of each of the tandem actuating cylinders in the flight control and stabilization systems.

PC 2 was an integrated power control and utility hydraulic system driven by two starter-pumps, one mounted on each left-hand power plant. The system provided hydraulic pressure to one-half of each of the tandem actuating cylinders in the flight control and stabilization systems. The system also supplied hydraulic pressure for the utility hydraulic system. The utility system, which provided hydraulic pressure for operation of the flaps, landing gear, wheel brakes, and wing tilt was hydraulically separated from PC 2 by a manually-operated isolation valve.

The emergency utility hydraulic system was powered by the APU starter-pump. The system was separated from the APU starter-pump by a manually-controlled solenoid-operated shuttle valve which was normally closed except when a utility system failure occurred.

The engine start system consisted of starter-pumps mounted on each powerplant and a starter-pump mounted on the APU. Hydraulic power delivered by the APU pump drove the powerplant starter-pumps.

FUNCTIONAL ANALYSIS

APU start circuit

Hydraulic pressure surges from the APU start accumulator were high enough to blow seals in the APU starter-pump. Due to the magnitude of these surges there was also some doubt as to the strength of the starter-pump casing. Therefore, APU air restarts were prohibited. A surge valve should be incorporated to reduce the surge, and the starter-pump casing strength should be determined and increased if necessary. (B 80)

Approximately 5 minutes of vigorous pumping by four people was required to pressurize the APU start accumulator to 3200 psi. The pump handle was oriented so that most of the pumping action had to be done by arm movement. The pump handle position was such that it would be impossible to pressurize the accumulator if cargo was located near the pump. The pump and handle should be positioned so that the cargo will not interfere with its operation and low enough to allow pumping action by arm and body movements. (D 81)

engine starter-pump

Numerous engine starter-pump units were replaced during Category I testing. The replacements were required because of internal wear which was detected by loss of power available to rotate the engine. A factor in the starter-pump wear was the continual requirement to cool the engines by motoring after shutdown. The starter-pump life should be improved. (C 83)

brake system

The braking system was inadequate as evidenced by overheating, fading, relatively long ground rolls, and the absence of deceleration effects. Ground roll distances are discussed in the Performance section. The braking system should be improved to increase effectiveness and decrease the landing ground roll. (C 22)

main landing gear

Free fall of the main gear was inadequate to obtain a positive downlock condition. Application of about 1 1/2 g's was required to insure positive action. A positive means of obtaining a downlock condition without applying a g load should be incorporated. (B 87)

The landing gear often appeared to the chase pilot to be in the up position when the warning circuit indicated an unlocked condition. During several ground retraction tests the warning circuit indicated that the gear was up and locked when it was not. A reliable gear position indication system should be incorporated. (C 88)

Emergency extension of the landing gear was accomplished by actuating manually-controlled valves in the cargo compartment. A control should be included in the cockpit to provide emergency landing gear extension. (C 48)

propeller transfer valves

Considerable leakage occurred from the main and tail propeller transfer valves under static and dynamic conditions because of inadequate seals. There was a leakage rate of one drop every 40 seconds in the static condition. It was estimated that the leakage rate during the dynamic condition was approximately one drop per 25 cycles of transfer valve operation. Leakage from these valves was ingested into the engines and resulted in a loss of power. This problem is also discussed in the Propulsion section. The transfer valve and IGC hydraulic leaks should be eliminated. (C 61)

nose gear

During hover, the nose gear cocked to one side due to a lack of a positive centering device. A bungee cord system was successfully used to center the nose gear. The nose gear on production aircraft should be designed with a positive centering device. (D 90)

During the first portion of the Category I program, a nosegear shimmy problem was encountered. The exact cause was not determined. The shimmy damper linkage was tightened and silicon fluid was used in the damper. These changes appeared to solve the problem.

wing tilt

Originally, the wing tilt and flap hydraulic systems were powered by PC 2 during normal operation or by APU hydraulic power for emergency operation. If one PC 2 pump failed, the maximum wing tilt rate was lowered to an unacceptable rate for emergency situations. The hydraulic system was modified to make APU power the main source of hydraulic pressure and PC 2 the backup source. This change provided the maximum wing tilt rate in case of a PC 2 pump failure. Production aircraft should have the capability of maximum wing tilt rate during all conditions. (D 85)

hydraulic fluid quantity indication

The hydraulic system contained no method for determining fluid quantity in flight. A fluid quantity indicator or a low-level warning system should be provided in the cockpit. (D 82)

power control designation

The hydraulic system designation did not conform with the standard numbering method. The PC 1 system was powered by hydraulic pumps on the right engines and the PC 2 system was powered by hydraulic pumps on the left engines. The cockpit indicators were located in the standard sequence with the PC 1 indicator to the left of the PC 2 indicator. The standard numbering sequence should be used for the hydraulic systems. (D 79)

cargo doors and ramp

The cargo doors and ramp could be operated only by utility hydraulic system pressure since there was no backup system. They could not be operated on the ground without using the APU, engines, or ground hydraulic power. A backup system should be incorporated on a production aircraft for rapid operation of the cargo doors and ramp under emergency conditions, and for ground operations when the APU and engines are not operating and ground hydraulic power is not available. (D 86)

HEATING AND VENTILATING SYSTEM

Ram air was used in the cockpit and cargo compartments as ventilating air in horizontal flight.

When ram air was not available, as in vertical flight or when on the ground, a ventilating air fan was energized through a pressure switch connected to the pitot static system. Airflow was controlled by manual regulation of dampers. Heating was provided by ventilating air heated by a 200 000-Btu combustion heater. A temperature control thermostat regulated air temperature. Safety provisions included pressure switches in the ventilation and combustion air ducts and an overtemperature thermostat.

- FUNCTIONAL ANALYSIS

heating

The combustion heater did not function properly as evidenced by an abnormal amount of smoke from the heater exhaust and some fuel drainage. This problem was attributed to an inefficient ventilating air fan. A temperature survey was not scheduled for Category I; therefore, quantitative data were not available. A complete temperature survey will be accomplished during the Category II program. The heating system should be improved on production aircraft. (D 91)

Only a limited amount of flying was accomplished at high altitudes. A qualitative analysis did not reveal any major discrepancies in the heating system although solar radiation ("greenhouse" effect) helped to maintain comfortable temperature.

AVIONICS

- DESCRIPTION

The communication system consisted of an intercommunication set and a UHF radio set with an automatic direction finder. Provisions were made to allow later installation of an IFF system, an HF-SSB radio, a Tacan system, and a VHF-FM radio.

The navigation systems included a radio compass, a VOR/LOC radio receiver, a glide slope receiver, a marker beacon receiver, a compass and gyro system, and a flight director system. A radar altimeter will be included for Category II.

- FUNCTIONAL ANALYSIS

communications

The UHF system used antennas installed on the top and bottom of the fuselage. There was so much noise and interference associated with the use of the bottom antenna that it was unusable. The cause for this discrepancy was not determined due to the low priority of such testing during the Category I program. Communications using the upper antenna were limited when the aircraft was headed toward the desired station with the wing in the up position. There was also a considerable amount of static in the UHF communication system at a 20 000-foot altitude. This was unexplained at the time of the MPP. These discrepancies should be further investigated and corrected. (D 93)

The intercommunications system was generally very noisy. This problem was being investigated by the contractor and should be corrected during the Category I test program. (C 92)

The radio compass was 180 degrees out of phase during ground check when the static ground wire was attached to the aircraft. The cause for this had not been determined at the time of the MPP.

navigation systems

It was planned that the cables for the radar altimeter would be stowed and the system would not be checked out prior to the aircraft delivery to Edwards AFB. No work was planned to verify the operation of the flight director computer. These systems should be functionally checked prior to aircraft delivery to Edwards AFB. (B 97)

AUXILIARY POWER UNIT

DESCRIPTION

The APU consisted of a Solar T-62T-19 gas turbine engine, a hydraulic starter-pump and a 25-kva generator. The APU system provided hydraulic power for ground system checkout, main engine starting, wing tilt, and power for the emergency utility hydraulic system during flight. The APU system also provided electrical power for the entire electrical system if the CSD generator failed.

The APU was started through the use of accumulator hydraulic pressure. The accumulator could be hand pumped and was automatically recharged during APU operation.

FUNCTIONAL ANALYSIS

APU overspeed

During several flights, as the aircraft approached a climb rate of 4000 feet per minute, the APU oversped and shut down. Efforts to stabilize the aircraft at various altitudes and allow the APU fuel control altitude compensator to react and restrict the fuel flow were unsuccessful. Trial and error adjustment of the minimum fuel flow adjustment established a fuel schedule that allowed for a 4000-foot per minute climb and sustained operation at altitudes up to 25 000 feet. The effects of this adjustment on the starting characteristics at altitude could not be evaluated until the APU hydraulic start circuit deficiencies were corrected. The APU airstarting capability and reliability should be demonstrated throughout the flight envelope. (B 98)

APU g loading sensitivity

The APU shut down on two separate occasions during hard landings. Additional testing was required to define the cause and sensitivity of the APU to high g loadings. This problem should be corrected. (D 99)

FLIGHT CONTROLS AND STABILIZATION SYSTEMS

DESCRIPTION

The flight controls consisted of those primary and secondary flight control systems necessary for aircraft control during cruise, transition, and vertical or hover modes of flight. The primary flight controls consisted of the lateral, directional, longitudinal, and height control systems. The secondary flight controls consisted of the wing incidence, leading edge slat, and trailing edge flap systems.

The stabilization system provided stability augmentation in pitch, roll, yaw, and height during VTOL, hover, and STOL flight modes. The trim system, which was part of the stabilization package, provided trim in pitch, roll, and yaw for all modes of flight.

FUNCTIONAL ANALYSIS

propeller control

The propeller rpm drifted because of a null shift in the governor servo valve and was proportional to temperature. The system should be modified to compensate for temperature. (B 100)

aileron servo valve

The aileron servo valve could be overstressed during ground operations when the wing was up, the flaps were down, and no hydraulic power was available to the flight control system. The problem was caused by the aileron programer stops which limited down aileron movement with the flaps extended. Without hydraulic power to the flight control system, the aileron programer stops caused the weight of the ailerons to be supported by the servo valve linkages. This condition overstressed the servo valve which could not be inspected during a normal preflight inspection. This problem should be corrected prior to Category II testing. (A 3)

slat warning system

The slat warning light illuminated frequently even though the slats were in their programed position. This light should illuminate whenever a slat does not move to its programed position. This problem should be corrected prior to the Category II program. (B 104)

main propeller beta control

The main propeller beta control linkage drifted under g loads which resulted in propeller rpm changes. This snould be corrected. (C 101)

stability augmentation system (SAS)

Consistent pitch trim "creep" on the ground before takeoff was attributed to tail propeller vibration. The pitch trim system should be modified so that the trim will not change unless actuated by the pilot. (C 102)

■ ICE PROTECTION

● DESCRIPTION

The anti-icing systems were designed to prevent ice from forming on the pitot heads, windshield, engines, and nacelles. The pitot head and windshield anti-icing systems used electrical power for operation. The engine and nacelle anti-icing system used engine compressor bleed air for operation.

The deicing systems provided ice protection to the main propellers, wing, and UHT. The propeller deicing system used electrical power for operation. The wing and tail deicing systems used compressor bleed air from the two left engines for inflation of pneumatic boots on the leading edge of the wing and horizontal tail.

The ice detection system was designed to use an ice detector probe in the air intake duct of engines No. 1 and 2. An engine ice caution light would illuminate when either of the two ice detector probes sensed the formation of ice.

● FUNCTIONAL ANALYSIS

engine nacelle anti-icing

Engine nacelle anti-icing was not scheduled for installation in any of the five aircraft. Engine nacelle anti-icing and all other ice protection equipment should be installed in aircraft No. 4 or 5 for evaluation during Category II. (C 105)

■ INSTRUMENTS

● DESCRIPTION

The flight instruments were primarily standard Air Force instruments and are described in the Cockpit Evaluation section.

● FUNCTIONAL ANALYSIS

instrument wear and damage

Due to the considerable amount of ground checkout time, many of the instruments for indicating torque, T_{t5}, fuel flow, and oil pressure were worn and had to be replaced. These instruments were powered by the emergency generator which was operating whenever the PC 2 hydraulic system was pressurized and the APU and CSD generators were inoperative. During the Category II test program, the appropriate circuit breakers should be pulled to prevent excessive operation times on these instruments. Production aircraft should have a master instrumentation switch. (D 41)

The fuel flow transducer was lubricated by the fuel, and therefore had to have fuel in it during installation. This precaution was not observed on several occasions and the transducer was damaged.

system checkout

Several instruments and systems were not tested. The total torque system was not tested because telemetry instrumentation was installed in its place. The radar altimeter was not tested because an additional airspeed indicator was installed in its place. The aircraft pitot-static system had not been checked without the instrumentation boom installed. These systems should be checked out prior to aircraft delivery to Edwards AFB. (D 96)

PERSONNEL SUBSYSTEM TEST AND EVALUATION

■ MANUALS

The original Flight and Maintenance manuals were outdated due to aircraft changes during early Category I tests and should be updated. (C 112)

Loading information will be required to properly load the aircraft during Category II operational suitability tests. A limited Cargo Loading Manual should be provided. (C 113)

■ STATIC ELECTRICITY

The appropriate manuals stressed the importance of danger from static electricity to ground personnel but there was no permanently attached static ground wire. A ground wire should be considered for production aircraft since the V/STOL operations concept is based on a minimum of ground support personnel and equipment. (D 111)

■ NOISE

● QUALITATIVE DATA

Qualitative data gathered from the military and contractor test pilots indicated that the noise levels produced by the XC-142A exceeded allowable and desired levels. The noise intensity was sufficient to cause vibrations of support equipment up to 300 feet from the aircraft during hover and STOL operations. Observers could also feel the pressure waves as vibrations against their bodies.

Pilots flying the aircraft were required to wear their helmets to adequately communicate with one another and with the ground using the microphone in their oxygen masks. They considered the headset within their standard helmets (P-1 and P-4 types) inadequate because of difficulty in understanding transmissions. It was believed that this difficulty was caused by sound pressure waves entering the space between the pilot's head and the helmet shell, and that this space then acted like a small echo chamber. The Protection, Inc., or Lombard-type helmet was helpful in reducing this communication problem. The Lombard-type helmet had a seal

around the helmet's perimeter and solid continuous padding which effectively reduced the echo chamber effect. The high noise levels were associated with the V/STOL flight regime and were reduced during cruise to a level comparable with other turboprop aircraft.

Contractor theoretical noise levels in the cargo compartment were higher than those in the cockpit. Since it was essential that the pilots used sealed-type helmets and closed mikes, crewmembers in the cargo compartment would require similar devices for communication. Ear protection would be essential for all crew and passenger personnel. With ear plugs and ear muffs, the maximum sound levels specified in MIL-A-8806 and AFSCM 160-3 would have been exceeded. Noise levels inside the aircraft could be reduced by proper insulation and closing of all air transfer openings, but it was estimated that the noise levels would still be in excess of the 120-db limit established in MIL-A-8806. The sound levels in the aircraft should be reduced and all flight crew, passenger, and ground crew personnel should be provided with suitable ear protection equipment. Allowable exposure limits should be determined. (C 107)

Some type of ear protection was necessary for ground personnel working within the aircraft's runup area. Air Force ear plugs were worn by some military personnel and seemed sufficient for normal runup. Military personnel reported extreme discomfort during engine runups if ear protection was removed. Ear muffs were considered adequate and better than plugs and should be the minimum mandatory requirement for runups. Fifty percent of the contractor ground personnel wore ear muffs of the Air Force type (sponge rubber with a liquid glycerine fill), while 20 percent wore Air Force ear plugs, and the rest wore no ear protection. During V/STOL operations ear plugs were considered to be inadequate.

QUANTITATIVE DATA

military data

For normal starts and ground checkout, the noise levels were as shown in figure 3.

Insufficient military data were available to completely check the contractor's takeoff power sound level chart in the Flight Manual (T.O. 1C-142(X)A-1). Enough military data were obtained; however, to indicate these sound levels can reasonably be expected. This chart is reproduced for convenience in figure 4

Ear protection is mandatory for personnel exposed to a sound pressure level of 100 decibels for any time period greater than 2 1/2 hours. Protection is recommended for any exposure time longer than 15 minutes. For a 110-decibel sound pressure level, 15 minutes is the maximum exposure time without ear protection, and protection is recommended for any exposure time exceeding 1 minute. For a sound pressure level of 120 decibels, the maximum exposure time without ear protection is 1 minute 30 seconds (AFR 160-3).

With only the APU running, a noise level of 94 decibels was registered within the cargo compartment just aft of the cockpit bulkhead. This noise increased 2 to 3 decibels while the wing was in rotation.

Sound levels were measured adjacent to the runway during the STOL tests. During these tests the wind speeds were 2 to 3 knots. The results are shown in table XIV. The data are not definitive due to a large amount of scatter. The noises were high and could present a problem in an occupied area.

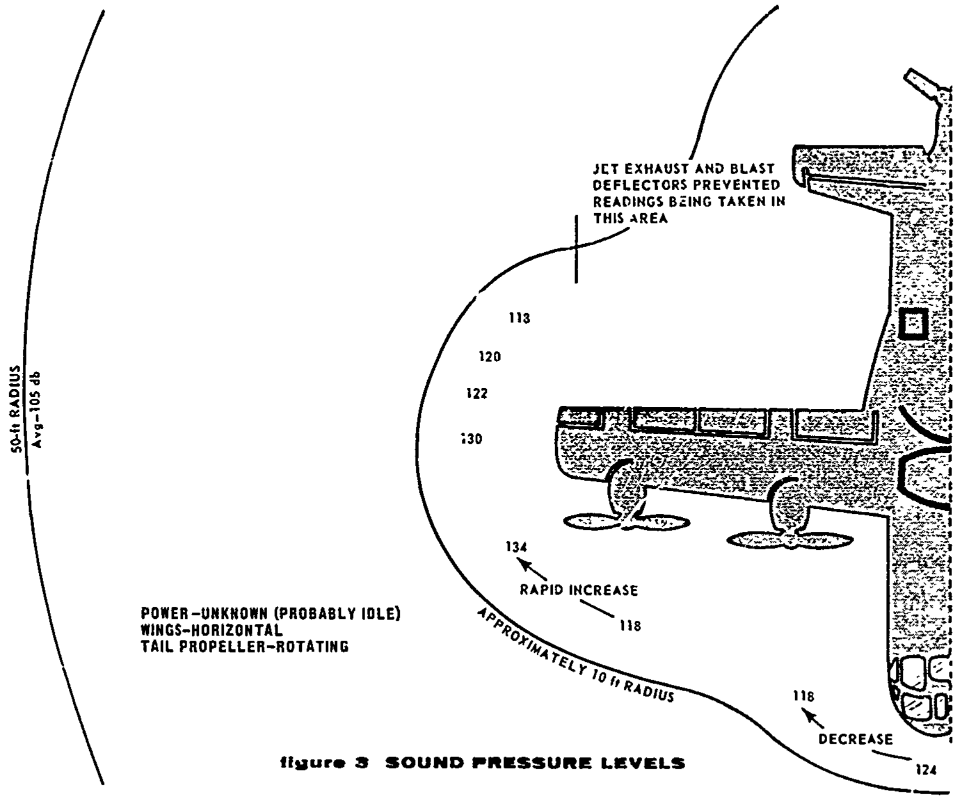
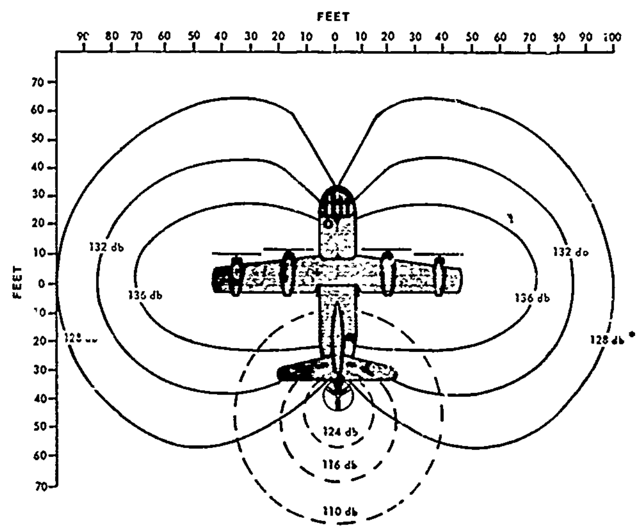


figure 3 SOUND PRESSURE LEVELS



Note

SOUND LEVELS SHOWN FOR FOUR ENGINES RUNNING AT TAKEOFF POWER. TWO-ENGINE OPERATION PRODUCES APPROXIMATELY THE SAME SOUND LEVELS.

TAIL PROPELLER SOUND LEVEL SHOWN FOR SOUND PRODUCED BY TAIL PROPELLER OPERATION ONLY.

EAR PROTECTION IS REQUIRED AT SOUND LEVELS ABOVE 90 db.

*122 db AT 140 ft
110 db AT 300 ft
100 db AT 500 ft
90 db AT 900 ft

figure 4 DANGER AREAS-NOISE

TABLE XIV

XC-142A STOL SOUND LEVELS

Takeoffs		Landings	
Estimated Slant Range from Aircraft (ft)	Sound Pressure Levels (db)	Estimated Slant Range from Aircraft (ft)	Sound Pressure Levels (db)
135	120+	195	120+
135	125+	135	125
235	115	135	130+
195	124	235	123
210	124	165	124+
75	128+	30	130+
260	118	80	124+

contractor data

The only known noise measurements from the interior of the XC-142A during actual flight were made during flight 28 on 15 Feb 1965 on the No. 2 aircraft. The recorded data are shown in table XV.

TABLE XV

XC-142A IN-FLIGHT CARGO COMPARTMENT SOUND LEVELS

Configuration	Flight Condition	Sound Pressure Levels (db)
Takeoff Power	STOL Takeoff	130
Total hp - 4949 Wing - 25 deg Flap - 60 deg Thrust - 17 602 lb	Descent at 45 KIAS	122
Total hp - 3700 Wing - 25 deg Flap - 60 deg	Descent at 45 KIAS	122
Not Recorded	Uncontrolled descent prior to 1800-ft pullout	126

Note: Measurements were taken 3 feet aft of the propeller plane near the centerline of the aircraft. Prop rpm was approximately 91 percent.

The contractor measured the sound levels in the center of the cargo compartment during tiedown runups for several simulated flight conditions. The values considered most reliable by the contractor are shown in table XVI. The actual measurements exceeded the theoretical values by 3 to 6 decibels.

TABLE XVI

CONTRACTOR NOISE RECORDING

Taken During Tiedown Runups

Simulated Condition	Actual Measurements (db)	Theoretical Sound Level (db)
Cruise	135 - 144	128 - 140
Conventional Takeoff	147 - 156	140 - 153
Short Takeoff	None	144 - 152
Vertical Takeoff	None	144 - 148

Note: Measurements were taken in the center of the cargo compartment. This summary was obtained from the Final Environmental Vibration and Acoustics Report, No. 2-53450/3R-880, dated 10 June 1964, provided by LTV to SPO.

No information was available on the possible noise attenuation due to the cargo compartment ballast tanks. These were large structures filled with various amounts of water to simulate various payloads.

The contractor noise tests and calculations were made using the sandwich-constructed panel (Metalite) selected as the production aircraft skin. This type panel was claimed to have a high acoustic transmission loss. This attenuation, combined with additional acoustic insulation and elimination of openings, should provide sufficient noise suppression to prevent any hearing loss. Actual effects of these various considerations must be tested before an evaluation can be made.

TEMPERATURE

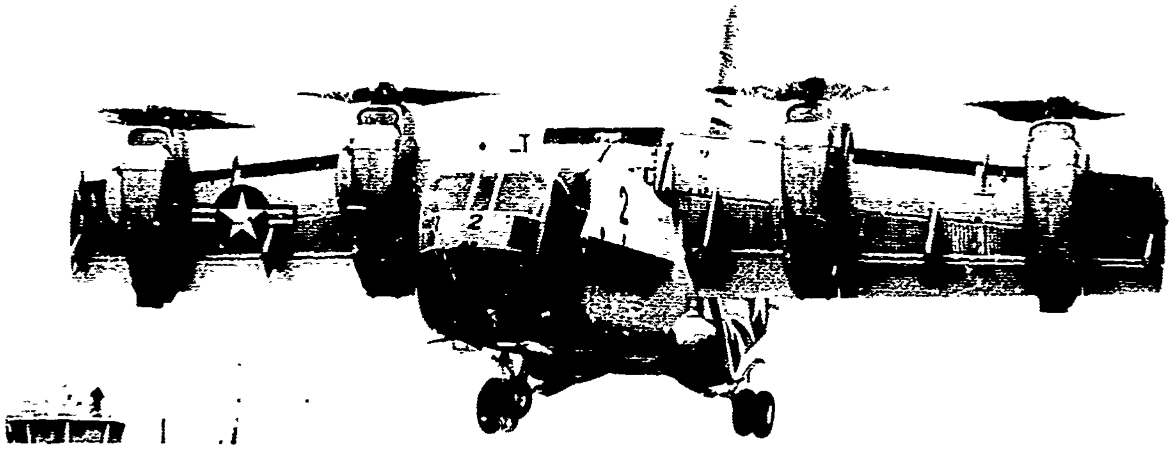
The cockpit temperatures were too high and the pilots were comfortable only when the ambient temperature was below zero degrees C.

This high temperature was caused by the lack of an adequate ventilation and blower system, and by the "greenhouse" effect of the plexiglass windshield. No quantitative data were collected but the pilots' estimates were that a temperature differential of about 35 degrees F existed between the interior air and the ambient air. The fan provided the only means for cooling and was inadequate. "Cool suits" should be provided when the aircraft are delivered for Category II. (B 108)

The tolerance for hot environments is much more critical than that for cold. If tolerance limits are exceeded in the hot environment an acute physiological condition with fainting and prostration may quickly ensue (AFSCM 80-1, C.6-2.3.1.1). Air-conditioning should be provided in production aircraft. (D 109)

WIND BLAST

Propeller wash was hazardous since high winds were created during V/STOL operations. This wind blast knocked down personnel and affected equipment located more than 75 feet from the aircraft. Loose objects would be a hazard to personnel, supplies, and/or equipment in the immediate area of V/STOL operations. The hazardous wind blast areas should be defined. (D 106)



CONCLUSIONS AND RECOMMENDATIONS

GENERAL

The XC-142A design objective was to provide a full scale tilt-wing V/STOL transport aircraft with which the operational capabilities could be determined for V/STOL aircraft in general and tilt-wing V/STOL aircraft in particular. It was not intended to be a production model. As a concept evaluation vehicle it was satisfactory except for three known safety of flight deficiencies and 22 known deficiencies which would interfere with Category II testing if not corrected. There were 44 additional deficiencies which should be corrected and re-evaluated during Category II tests and 44 more which should be corrected for a production C-142. Most of the known deficiencies were with systems which had not been sufficiently checked out before installation due to the limited funds available. The novel and critical systems (i.e., flight control system, gearboxes, cross-shafting, wing tilt, etc.) were more completely developed before installation and gave little trouble.

The aircraft was safe and simple to fly. Most of the deficiencies were in the aircraft subsystems. With the correction of 25 items the XC-142A would be ready for Category II tests. The XC-142A was not ready for production.

Each recommendation has the letter A, B, C, or D as a prefix. These letters denote the following:

A- Safety of Flight. Mandatory correction prior to delivery of the first XC-142A for Category II testing.

B- Deficiencies which will interfere with the Category II concept evaluation unless corrected. Mandatory correction prior to delivery of the first XC-142A for Category II testing.

C- Deficiencies which will not interfere with the Category II concept evaluation, but the corrections should be evaluated before the end of Category II. Mandatory correction before the end of Category I testing.

D-1 Deficiencies which should be corrected for an operational aircraft. Desirable correction before the end of Category I.

The following ASD comment applies to all recommendations preceded by an asterisk (*).

ASD COMMENT: ASD does not consider this item to be a requirement of the present program. Items will be reconsidered when and if a production configuration is established.

SAFETY OF FLIGHT DEFICIENCIES

The overhead emergency escape hatch handles were pinned to prevent them from vibrating loose in flight. The pins had to be pulled before the handles could be moved to release the hatches.

A 1. The overhead escape hatches should be secured so that they can be rapidly removed in emergency situations. The removal must be easily accomplished with only one "lock-unlock" control as specified in AFSCM 80-1 (C.6-2.4.3.3) (page 4).

ASD COMMENT: ASD concurs. All pins have been removed and easy single "lock-unlock" feature incorporated. Force requirements of AFSCM 80-1 have not been met.

Wing and fuselage shaft bearing temperatures could not be monitored in flight. Monitoring is necessary to warn of an impending bearing failure.

A 2. A wing and fuselage shaft bearing overheat warning system should be installed so that the pilots can detect an overheating condition in flight (page 37).

ASD COMMENT: ASD concurs and the recommendation has been incorporated.

The aileron servo valve could be overstressed during ground operations when the wing was up, the flaps down, and no hydraulic power was available to the flight control system.

A 3. The aileron control system should be modified to prevent damage to the aileron servo valve during ground operations (page 45).

ASD COMMENT: ASD concurs and all aircraft have been modified to comply with recommendation.

FLIGHT RESTRICTIONS

The design center of gravity limits were 15- to 28-percent MGC, but the range available for evaluation was only 17- to 28-percent MGC. The flight envelope limits during this evaluation were too restrictive to adequately evaluate the XC-142A maneuvering characteristics. The rear cargo doors had not been cleared for in-flight operation. In the V/STOL flight regime the allowable flight limits had not been completely defined. The maximum translational speeds in hover had not been explored to either the design limits or the actual safe aircraft capability limits. Recirculation ground effects in hover required an increase in control activity. The limits of safe control activity and attitude had not been defined.

C 4. The structural limits should be expanded to 300 KEAS and 2.4 g's (page 25).

ASD COMMENT: ASD concurs, expect this capability prior to completion of Category I testing.

C 5. The center of gravity limits should be determined (page 15).

ASD COMMENT: ASD concurs. Center of gravity limits to be determined in Category I.

C 6. Stalls in the cruise configuration and a 10/30 degree CA landing configuration should be demonstrated (page 26).

ASD COMMENT: ASD concurs. Item complete.

C 7. The maximum safe sideslip limits should be determined (page 25).

ASD COMMENT: ASD concurs. This will be accomplished in Category I.

C 8. The maximum safe roll rate limits should be determined (page 26).

ASD COMMENT: ASD concurs. Item complete.

C 9. The rear cargo door operation should be demonstrated in flight at the maximum allowable speed with and without the tail propeller operating, and in hover (page 4).

ASD COMMENT: ASD concurs. This will be accomplished in Category I.

C 10. The flight limits should be determined for various SAS failure conditions (page 29).

ASD COMMENT: ASD concurs. Flight limits will be determined during Category I.

C 11. The hover translation envelope limits should be established (page 13).

ASD COMMENT: ASD concurs. Item complete.

C 12. The hover control and attitude limits should be determined (page 11).

ASD COMMENT: ASD concurs. Item complete.

*D 13. The structural envelope should be expanded to V_{max} at 3 g's (page 25).

PERFORMANCE AND STABILITY AND CONTROL (HANDLING QUALITIES)

Vertical takeoff and hover were easy to accomplish, although a possible recirculation caused turbulence between 4 and 10 feet above the ground. Controllability and maneuverability were good about all axes, and precise control was easily accomplished in smooth air with minor control motions. Height control was good if sufficient excess engine power was available. The aircraft was roll sensitive to gusts. The hover performance was less than predicted by about 12 percent. This was attributed to a loss in propeller efficiency and inadequate engine topping procedures. No change in power required to hover within ground effect was detected.

C 14. The propeller efficiency should be improved and the engine topping procedures changed to attain at least design hover performance (pages 12 and 37).

ASD COMMENT: ASD concurs. A redesigned propeller is being prepared for test. Engine topping procedures have been improved.

The directional control force gradients in the STOL, VTOL, and hover configurations were considered high by the pilots and exceeded the requirements of AGARD

408 and MIL-H-8501A. The contractor successfully demonstrated the hover handling characteristics with various SAS combinations, including all SAS off. Pilot proficiency in hover techniques was essential for safe operation with the SAS off. Roll SAS off characteristics were the most difficult from the pilot standpoint. The contractor successfully demonstrated a complete verticircuit with all SAS off. Care had to be exercised in roll to avoid high roll rates. Moderate directional disturbances occurred near the ground.

C 15. The rudder force gradient should be decreased by 50 percent in the STOL and VTOL configurations (page 10).

ASD COMMENT: A lower force gradient is being designed for this configuration.

Conversions were easily accomplished and could be initiated from hover or climb. Trim changes during conversions were negligible. Except for a slight decrease in roll damping there was little deterioration in flying qualities during conversions with one SAS channel inoperative on each axis.

The aircraft acceleration was outstanding, accelerating to a 40 KIAS lift-off speed in 3.8 seconds. Controllability about the roll and pitch axes was good. The static directional stability was unsatisfactory, and precise directional control was difficult, especially at wing angles above 35 degrees. Trim changes during takeoff and climbout were negligible. Crosswind takeoff and landing characteristics were satisfactory.

C 16. The STOL static directional stability should be improved (page 15).

ASD COMMENT: ASD concurs. The item is under study for further flight test.

Maximum power climb performance was outstanding for a cargo/transport-type aircraft. At a gross weight of 35 900 pounds, the time to climb from sea level to 25 000 feet was 6.1 minutes. During maximum power climbs, the aircraft was sensitive longitudinally and attention was required for airspeed control. Persistent small yaw excursions of less than 2 degrees were observed throughout the climbs.

The best specific range was obtained with two engines operating. Specific range with two engines operating was 4 to 12 percent less than predicted in the Flight Manual. The best cruise speeds were very close to predictions. There was no noticeable difference in total power required between 2- and 4-engine cruise at the same conditions.

The longitudinal stick force per g was tested at 210 KCAS and was too low. Although it met the design requirement based on a class III fighter aircraft (refer to MIL-F-8785(ASG)), it did not meet the class II cargo aircraft requirement.

B 17. The longitudinal stick force gradient in cruise should be increased to meet MIL-F-8785(ASG) class II cargo requirements (page 25).

ASD COMMENT: ASD does not concur. A study is being made to determine the feasibility of optimizing the system over the range of airspeeds and altitudes of a V/STOL aircraft.

The rudder deflection, force required, and the static directional stability were low at cruise speeds.

Although the rudder pedal forces were within the limits of MIL-F-8785(ASG), they were considered unsatisfactory when coupled with the low static directional stability in cruise.

C 18. The rudder forces in cruise should be increased (page 25).

ASD COMMENT: ASD concurs. New hardware is being designed.

"Loping" was manifested in the cockpit by longitudinal acceleration pulses of small magnitude and varying frequency. It occurred during high-speed low-power clean configuration descents. The contractor attributed this to the propeller governor operating at the minimum governing rpm.

*D 19. The "loping" during high-speed low-power descents should be investigated and corrected (page 19).

Buffet onset and buildup characteristics were similar for all configurations tested, and were very close to predictions. The light buffet region was adequate to warn the pilot during descents. The buffet onset speed was too high in the cruise configuration since it was coincident with the best climb speed. At CA's of 14/30 and 0/0 degrees the buffet onset speeds were 6 [illegible] 25 KIAS, respectively, above the minimum usable airspeeds. Recovery from descent bounday buffet was easily made by power application.

*C 20. The buffet onset speed at a CA of 0/0 degrees should be reduced (page 26).

The reconversion and vertical landing were easily accomplished and could be made any time there was sufficient power to hover. Although the flight capabilities for pattern and field operations were very flexible, the procedure for setting up the landing configuration was much more complicated than that for a conventional aircraft. Approaches were easily made to a specific point for hover or vertical landing. A slight floating or ballooning tendency was present at the lower wing angles if the wing incidence was changed too rapidly, but trim changes were negligible throughout the entire reconversion. A minimum height of 50 feet had to be maintained during reconversion to avoid ground effect instabilities at high wing angles.

The landing distances over a 50-foot obstacle were too long for an aircraft of this type and were not compatible with the short takeoff distances. In all cases tested the landing distances were longer than predicted. The brakes overheated and faded during taxi. During maximum performance landings the brakes were inadequate as evidenced by the relatively long ground rolls and the absence of high deceleration effects.

*C 21. The short landing performance should be improved to be compatible with the takeoff performance (page 22).

C 22. The braking system should be improved (pages 9, 21, and 41).

ASD COMMENT: ASD concurs. Brake system is being improved.

During landing at wing angles of 30 to 35 degrees, there was a slight negative ground effect below about 10 feet. Directional control during landing deteriorated with increased wing angles, but was satisfactory up to 35 degrees. Serious stability and control deterioration occurred during a landing with a CA of 45/60 degrees, and to a lesser extent during a takeoff with a CA of 45/30 degrees.

The effects of flap program changes were not evaluated. The aircraft had good waveoff characteristics.

C 23. The STOL CA limits should be determined (pages 15 and 22).

ASD COMMENT: ASD concurs. This will be accomplished during Category I.

SYSTEMS EVALUATION

AIRFRAME

The plexiglass windshields exhibited a low-frequency vibration at airspeeds above 215 KIAS which was annoying to the pilots and potentially dangerous. They were not impact-proof.

C 24. The strength of the windshields should be reevaluated, and the effects of the vibrations on their fatigue life should be determined. The cause of the vibrations should be determined and the vibrations should be eliminated (page 30).

ASD COMMENT: Vibration effects have been evaluated. Stresses on glass and structures are low. No further action is contemplated for present program.

*D 25. The windshields should be made impact-proof (page 30).

The wing used for static structural tests failed twice at about 110 percent of the limit load for a 50-fps gust at 350 KEAS. Wing ribs cracked on the No. 1 aircraft during the 50-hour tiedown tests. Redesigned ribs were installed on the No. 1 aircraft wing and the static test wing.

C 26. The modified wing ribs should be installed on all XC-142A aircraft (page 31).

ASD COMMENT: ASD concurs. Redesigned ribs have been installed on all aircraft.

*D 27. The static test wing should be modified and tested to 150 percent of limit load (page 31).

The personnel entrance door opened inadvertently in flight which was attributed to lack of an overcentering lock. An escape hatch tore loose and was lost in flight. This failure was attributed to a weak hinge point. Tail propeller shaft failure could result in damage to the fuselage shaft tunnel and to the fuel cells. The front air deflector did not always remain properly sequenced to the wing. This resulted in the wing being lowered over, and damaging, the deflector.

B 28. The personnel entrance door lock should include an overcentering mechanism to insure positive locking (page 31).

ASD COMMENT: ASD concurs. All aircraft have been modified.

B 29. A reliable system should be incorporated which will insure the proper sequencing of the forward air deflector with the wing (page 31).

ASD COMMENT: ASD concurs. All aircraft have been modified.

C 30. The escape hatch hinge point should be strengthened to withstand airspeeds through V_{max} (page 30).

ASD COMMENT: ASD concurs. All aircraft have been modified.

*C 31. Shielding should be provided in the fuel cell area to prevent possible fuel cell rupture by a shaft failure (page 31).

The top of the cargo door (in the up position) was used by maintenance personnel as a walkway. This surface had no protection. Many access panels were attached with screws rather than Dzus or camlock fasteners.

C 32. The top of the upper cargo door should have a walkway if it does not interfere with the airflow, or a protective cover should be provided (page 31).

ASD COMMENT: ASD does not concur. These are "no step" areas.

*D 33. Nonstress access panels should be attached with Dzus or camlock-type fasteners (page 31).

The overhead plexiglass panels directly above the pilot and copilot stations contributed to heat buildup because of solar radiation ("greenhouse effect"). They had been painted over to reduce the temperature increase, which could be as much as 35 degrees F above ambient. Overhead visbility was required to clear traffic before a vertical takeoff.

*D 34. Overhead windows should be installed in V/STOL aircraft cockpits. Adjustable sunglare curtains should be installed to decrease the solar radiation effect (pages 6 and 30).

A 3-step stand was normally used by the pilots to enter the aircraft, but this was not aircraft equipment. No hard points specifically for snatch pulleys were provided to help in loading cargo. No winch was provided to pull cargo into the cargo compartment. The fuselage jacks used for the aircraft were not found in the Air Force inventory since they had to be short yet have a long stroke.

B 35. The required jacks should be provided with the aircraft (page 32).

ASD COMMENT: ASD concurs. Jacks have been delivered.

*D 36. Suitable lightweight steps should be provided (page 4).

*D 37. A winch should be provided to facilitate loading cargo (page 32).

*D 38. If existing cargo tiedowns are unsuitable for use as snatch pulley attachment points, then hard points should be provided (page 32).

COCKPIT

Except for the vertical direction, ground and air visibility from the cockpit was excellent.

The effectiveness of several of the cockpit instruments was degraded by their locations and/or markings. Many instruments were worn and had to be replaced due to the considerable amount of aircraft ground checkout time. The engine oil pressure indicators were located on the center panel. The gearbox oil pressure indicators and engine/gearbox oil temperature indicators were on the overhead panel where they were difficult to read.

B 39. The wing/flap position indicators should be graduated in 5-degree increments and numbered every 10 degrees. They should be relocated near the flight instruments to conform to AFSCM 80-1 (C.2-2.2.2.6.3) (page 6).

ASD COMMENT: ASD concurs. All aircraft have been modified.

C 40. The engine and gearbox oil pressure indicators should be combined in a dual needle instrument. The engine and gearbox oil temperature indicators should be similarly combined. These pressure and temperature instruments should be located on the center panel (page 6).

ASD COMMENT: ASD does not concur; too costly for present program.

*D 41. A master instrumentation switch should be provided to reduce unnecessary instrument operating time (page 46).

*D 42. The standby magnetic compass should be relocated so that it will not impair visibility from the cockpit (page 6).

*D 43. The total torque indicator should be deleted (page 6).

*D 44. The overhead panel should be inclined more and shifted forward (page 7).

The effectiveness of several of the controls and switches was degraded due to their location. For example, the emergency landing gear extension was accomplished only by manually-controlled valves in the cargo compartment. The collective lever had to be raised slightly during short landings to allow retraction of the lower collective gate solenoid. This delayed attaining ground idle and lengthened the ground roll.

B 45. The lower collective gate solenoid should be modified to provide only increased friction force with no positive stop (page 8).

ASD COMMENT: ASD concurs; the item is under study.

C 46. The emergency wing control switch on the overhead panel should be relocated adjacent to the primary-secondary switch (page 7).

ASD COMMENT: ASD does not concur. All emergency switches are maintained in a common location.

*C 47. The propeller governor switch should be relocated or duplicated to permit actuation by either pilot with his shoulder harness locked (page 7).

C 48. A control should be included in the cockpit to provide emergency landing gear extension (page 41).

ASD COMMENT: ASD concurs. This will be accomplished during Category I.

*D 49. The APU fire control T-handle should be changed to allow immediate physical distinction from adjacent engine fire control handles (page 6).

*D 50. The chip-detector wafer switch should be relocated for easier access and to conform to AFSCM 80-1 (C.2-1.3.1) (page 7).

*D 51. The landing gear handle should be relocated or duplicated to permit actuation by either pilot with his shoulder harness locked (page 6).

Ejection seats were provided in the five test aircraft, but were not planned for production aircraft. They were provided with height adjustments but not fore and aft adjustments.

*D 52. Zero-zero ejection seats should be provided in production V/STOL aircraft for the pilots (page 6).

*D 53. Fore and aft pilot seat adjustment capability as recommended in AFSCM 80-1, Volume III, drawing AD3, should be provided (page 6).

Instrument flight simulation equipment was not provided.

C 54. IFR simulation equipment should be provided so that the instrument flight capabilities of the XC-142A can be evaluated (page 8).

ASD COMMENT: AFFTC agreed to accomplish this task.

ENGINES

Engine failure had no effect on handling qualities when the cross-shafting was operating. Engine starting times were almost double the indicated Flight Manual time of 35 seconds. Engine front frame cracks caused removal of three engines during the propulsion integrated test stand (PITS) tests. The FOD rate on the T64-GE-1 engine was very high.

B 55. The engine front frames should be modified to prevent cracking (page 35).

ASD COMMENT: ASD concurs. A modified frame is being tested and will be installed if necessary.

*C 56. The compressor inlet should be redesigned and/or a separator should be considered to reduce the FOD (page 34).

*D 57. The engine starting times should be reduced to 35 seconds or less (pages 8 and 33).

The engine and IGC oil tanks cracked and leaked early in the program. The mounting straps frequently broke and it was difficult to install the tanks using the straps. When an engine was shut down in flight, the power turbine windmilled but the gas generator turbine did not rotate, apparently due to the drag of the starter-pump. Since lubrication for both turbines was provided by the gas generator turbine pump, the power turbine lubrication was questionable. Hydraulic fluid leaked from the transfer valves and IGC's. The fluid accumulated in the engine and caused significant power losses. The technique used to determine the degree of power loss was time consuming and complex.

B 58. The engine and IGC oil tanks should have load pads to prevent cracking (page 36).

ASD COMMENT: ASD concurs. Improved mounting arrangements have been incorporated on all aircraft.

B 59. Installation of the engine and IGC oil tanks should be simplified (page 36).

ASD COMMENT: ASD concurs. Tanks have been modified.

C 60. Studies should be made to determine the adequacy of the power turbine lubrication with no gas generator turbine rpm. Operating restrictions should be established (page 33).

ASD COMMENT: ASD concurs. Time limitation of 1 1/2 hours incorporated in Flight Manual.

C 61. The transfer valve and IGC hydraulic leaks should be eliminated (pages 35 and 42).

ASD COMMENT: ASD concurs. Improvements are in progress.

Proper engine oil service levels were shown by indicator lights, but were deactivated when the engine switches were turned on. No oil level indication was available during flight.

*D 62. The engine oil level warning system should be changed to provide low level indications during flight, in addition to proper service levels before engine start (page 7).

POWER TRANSMISSION SYSTEM

The tridirectional gearcase was limited to a 10-hour inspection interval due to the failure of the input gear caused by a resonant frequency vibration associated with rpm. Inspections revealed abnormal wear to the wing shaft bearings. Many fuselage shaft bearings were practically inaccessible for inspection. Installation tolerances of the IGC and engines were very critical.

B 63. A modified input gear should be developed and incorporated in the tridirectional gearcase to eliminate fatigue failure and permit an increased inspection interval (page 36).

ASD COMMENT: ASD concurs. New gear has been developed and is being installed in the aircraft.

C 64. The wing shaft bearing life should be improved (page 37).

ASD COMMENT: ASD concurs. New designed bearings are being incorporated at overhaul.

*D 65. A complete maintainability program should be pursued to insure that shaft bearings and all other components are accessible for inspection (page 37).

*D 66. Precise installation procedures and techniques should be provided to insure that proper installation tolerances are maintained (page 38).

PROPELLERS

The contractor successfully demonstrated the feather operation of the No. 1 and 4 propellers. The aircraft was safely landed in this configuration with a 10/30 degree CA.

The contractor successfully demonstrated the feather operation of the No. 3 propeller. The aircraft was safely landed in this configuration with a 10/30 degree CA.

The hub moment produced by the main inboard propeller blades was found to be higher than expected which resulted in cracks in one IGC.

B 67. All main propeller hubs should be modified to compensate for the high stresses (page 37).

ASD COMMENT: ASD concurs. Propeller hubs have been redesigned and will be installed at overhaul.

FUEL SYSTEM

A defueling check valve had to be manually positioned inside the aircraft before defueling.

B 68. The defueling valve should be located with the single-point refueling controls. If this is not possible, a placard should be placed near the single-point refueling controls to remind ground personnel to open the valve (page 38).

ASD COMMENT: ASD concurs. A placard has been installed in the aircraft.

ELECTRICAL SYSTEM

The tail propeller clutch system used a time-delay feature which was influenced by transient currents and could result in premature application of full hydraulic system pressure. It was also possible to actuate the switches in the wrong sequence. Both of these situations occurred, and each time the tail propeller shaft failed. The propeller feather circuit required that the decouple switches for No. 2 and 3 engines be energized before activation of the feathering switches for propellers No. 1 and 4.

C 69. A reliable and safe tail propeller clutch control system, should be installed on all aircraft. A tail propeller "ON-SPEED" indicator should be considered (pages 7 and 39).

ASD COMMENT: ASD concurs. The system has been redesigned for greater reliability.

D 70. The decoupling operation should be automatic and integrated with the appropriate feathering switches, but the separate decoupling capability should be retained (pages 7 and 40).

ASD COMMENT: ASD concurs. Item complete.

*D 71. The feathering circuit should be designed so that it can be checked during preflight (page 40).

Transient currents in the two main electrical power source circuits resulted in undesired component operation.

*D 72. The electrical circuits in a production aircraft should be designed to prevent transient currents. (page 39).

The high vibration environment resulted in considerable wire breakages in the tachometer generator for the main propeller monitor, the propeller seizure warning circuit, and the SAS.

B 73. Wire breakages should be eliminated in the tachometer generator for the propeller governor, propeller seizure warning, and the SAS circuits (page 39).

ASD COMMENT: ASD concurs. This problem is partially corrected and is still under study.

The battery could only be used to provide ignition to start the APU.

*D 74. The battery should be connected to the electrical bus system for emergency lighting, intercom, and alarm system as specified in AFSCM 80-1 (C.6-2.4.1.4.4) (pages 7 and 40).

The wing caution light was on when the utility hydraulic control handle was in the ON/WING UNLOCKED position. Since this was the normal case during takeoff and landing, the light caused confusion. The solitary caution light for gearcase oil pressure illuminated and remained on when a propeller was feathered. Th re- vented monitoring the re ining operating gearcase oil pressures. The external lighting did not include a searchlight.

C 75. The wing caution circuit should be changed so it will not be illuminated under normal operating conditions (page 8).

ASD COMMENT: ASD concurs. This item has been corrected.

*D 76. The gearcase oil pressure warning circuit should be changed to deactivate the warning circuit of an inoperative gearcase (pages 7 and 40).

*D 77. A controllable searchlight(s) should be provided (page 40).

The existing flight test instrumentation was apparently adequate for Category I c ft development requirem ut would be inadequate to accurately define specific XC-142A characteristics in Category II.

B 78. The flight test instrumentation on the XC-142A's to be used for Category II performance, stability and control tests should be improved to provide adequate data (page 27).

ASD COMMENT: ASD concurs. Improved instrumentation has been incorporated on aircraft No. 1 and 3.

The emergency generator was operating and the instruments were on whenever the PC 2 system was operating if the CSD and APU generators were shut down.

HYDRAULIC SYSTEM

The power control (PC) systems numbering sequence was confusing in that the PC 1 system was powered by pumps on the right-hand engines and the PC 2 system was powered by pumps on the left-hand engines. The cockpit indicators were located in the standard sequence with the PC 1 indicator to the left of the PC 2 indicator.

*D 79. The PC systems should be renumbered so that PC 1 is powered by the left engines and PC 2 is powered by the right engines. The PC 1 cockpit indicator should remain to the left of the PC 2 indicator (page 42).

Hydraulic pressure surges from the APU start accumulator were high enough to possibly damage the APU starter-pump and prohibited APU airstarts. The APU start accumulator hand pump position was

such that it would be impossible to pressurize the accumulator if cargo was located next to the pump.

B 80. An APU start accumulator surge valve should be incorporated to reduce surge pressures, and the starter-pump casing strength should be determined and increased if necessary (page 41).

ASD COMMENT: ASD concurs. This system has been modified.

*D 81. The APU start accumulator pump and handle should be positioned so that the cargo will not interfere with its operation and low enough to allow pumping action by arm and body movements (page 41).

Hydraulic fluid quantity could not be checked in flight.

D 82. A hydraulic fluid quantity indicator or a low level warning system should be provided in the cockpit (pages 7 and 42).

ASD COMMENT: ASD concurs. Contractor proposal is under consideration.

Numerous engine starter-pump units were replaced due to internal wear.

C 83. The engine starter-pump life should be improved (page 41).

ASD COMMENT: ASD concurs. Latest improvements show increasing reliability.

The oil cooler door position light illuminated when one of the four doors was open, instead of an annunciator panel indication of an out-of-sequence condition.

*D 84. The oil cooler door warning should be in the form of an annunciator readout to indicate an out-of-sequence condition (page 7).

The maximum wing tilt rate was originally lowered to an unacceptable rate if one of the PC 2 pumps failed. The hydraulic system was modified to make APU power the main source of hydraulic pressure for wing tilt, and the PC 2 system the backup system.

*D 85. Maximum wing tilt rate capability should be provided during all flight conditions (page 42).

The cargo doors and ramp were powered from the utility hydraulic system, but no backup system was provided.

D 86. A backup system should be incorporated for rapid operation of the cargo doors and ramp under emergency conditions, and for ground operations when the APU and engines are not operating and ground hydraulic power is not available (page 42).

ASD COMMENT: ASD concurs. An evaluation is being made.

LANDING GEAR

Free fall of the main landing gear was not always adequate to obtain a positive downlock, and the landing gear position indication system was unreliable.

B 87. A positive means of obtaining a landing gear downlock condition without applying a g load should be incorporated (page 41).

ASD COMMENT: ASD concurs. Positive downlock provisions are being incorporated on all aircraft.

C 88. A reliable landing gear indication system should be incorporated (page 41).

ASD COMMENT: ASD concurs. The system has been improved.

The nosegear steering was activated by an ON-OFF switch on the control stick. It was seldom needed since sufficient directional control for normal taxi maneuvers was provided by differential propeller beta. The downwash caused the nosegear to turn to one side during hover.

C 89. The nosegear steering switch should be modified to provide steering only when pressed (page 9).

ASD COMMENT: ASD concurs. The contractor has been requested to submit a proposal.

*D 90. The nosegear should be designed with a positive centering device so that it is not turned by the downwash (page 42).

HEATING SYSTEM

The heating system used to heat the cockpit and cargo compartment did not function properly.

D 91. An improved heating system should be provided (page 43).

ASD COMMENT: ASD concurs. The system will be made to operate according to design requirements.

AVIONICS

The intercommunications system was too noisy. The UHF antenna on the underside of the fuselage could not be used due to noise and interference which could not be explained or corrected during the evaluation.

C 92. The intercommunications noise problem should be investigated and corrected (page 43).

ASD COMMENT: ASD concurs. The item is under study.

D 93. The noise and interference problem associated with use of the bottom UHF antenna should be investigated and corrected (page 43).

ASD COMMENT: ASD concurs. The item is under study.

All of the avionics equipment was not available for evaluation. There was no "hot mike" feature in the communications system, which would have been useful for interphone communication.

*C 94. An AIC-18 type "hot mike" should be provided (page 8).

*D 95. The balance of the radio and navigation aids should be evaluated as soon as practical by the test team (page 8).

Several instruments and their systems were not installed and could not be evaluated. For example, the cables for the radar altimeter were to be stowed and the system was not scheduled for checkout prior to aircraft delivery for Category II testing.

B 96. All instruments planned for the XC-142A should be installed and func-

tionally checked prior to delivery for Category II (page 46).

ASD COMMENT: ASD concurs. Instruments will be installed by the contractor and calibrated by AFFTC.

B 97. The radar altimeter and flight director computer should be functionally checked prior to aircraft delivery for Category II (page 44).

ASD COMMENT: ASD concurs. Instruments will be installed by the contractor and calibrated by AFFTC.

AUXILIARY POWER UNIT (APU)

The APU would overspeed and shut down at aircraft climb rates above 4000 fpm. Effects of the fuel control adjustments to compensate for altitude change could not be evaluated due to the airstart restriction. The APU shut down several times during hard landings, apparently due to g sensitivity.

B 98. The APU airstarting capability and reliability should be demonstrated throughout the flight envelope (page 44).

ASD COMMENT: ASD concurs. Study of a new fuel control installation is under way.

D 99. The APU g loading sensitivity should be corrected (page 44).

ASD COMMENT: ASD concurs. Improvements have been made. Further study is required.

FLIGHT CONTROLS

Most missions could be completed by using the beta backup procedure in case of a propeller governor failure. The 75-percent-minimum propeller speed could not always be attained in flight due to cold-soaking of the propeller governor. The propeller rpm drifted due to a null shift of the servo valve rpm output. The drift was proportional to temperature. The drift at times was enough to prevent obtaining the required thrust. The main propeller beta control linkage was affected by g loads.

B 100. The propeller controls should be modified to compensate for temperature (pages 16 and 44).

ASD COMMENT: ASD concurs. The system has been modified.

C 101. The main propeller beta control system should be corrected so that g loads do not affect propeller rpm (page 45).

ASD COMMENT: ASD concurs. All aircraft have been modified.

Consistent pitch trim "creep" on the ground before takeoff was attributed to tail propeller vibration.

C 102. The pitch trim system should be modified so that the trim does not change unless actuated by the pilot (page 45).

ASD COMMENT: ASD concurs. All aircraft have been modified.

The engine topping and collective stick rigging procedures did not allow the pilot to exceed takeoff power on the remaining engines for recovery if an engine failed in hover.

C 103. The collective stick and engines should be rigged so that takeoff power

can be exceeded by pulling through the upper collective gate if an engine failure or other emergency occurs during VTOL or hover (page 33).

ASD COMMENT: ASD concurs. The item is under study.

The slat warning light frequently gave erroneous indications.

B 104. A reliable slat warning circuit should be provided (page 45).

ASD COMMENT: ASD concurs. The item has been corrected.

ICE PROTECTION

Engine nacelle anti-icing equipment was not scheduled for installation on any of the five XC-142A aircraft.

C 105. Engine nacelle anti-icing and all other ice protection equipment should be installed in aircraft No. 4 or 5 for evaluation during Category II (page 45).

ASD COMMENT: ASD concurs. No. 5 aircraft will be delivered with equipment required by the detail specification. No 4 will be retrofit.

PERSONNEL SUBSYSTEMS TEST AND EVALUATION

PROPELLER WIND BLAST

Propeller wind blast was a hazard to personnel and equipment during VTOL, STOL, hover, and high power engine ground runups.

*D 106. Wind blast hazard areas should be adequately defined (page 50).

NOISE

Noise levels were high in and adjacent to the XC-142A during high power operations in the V/STOL flight regime. The sound pressure level limits in MIL-A-8806 and AFSCM 160-3 were exceeded. The sound levels were high enough to cause hearing loss even with normal ear protection equipment. Lombard-type helmets were required by the pilots for noise suppression because the P-1 and P-4 type helmets were inadequate.

C 107. Sound levels in the aircraft should be reduced and all crew, passenger, and ground personnel should be provided with suitable ear protection equipment. The problem should be studied to determine the best methods for sound suppression and allowable exposure limits (page 46).

ASD COMMENT: ASD concurs. This item is under study.

CREW COMFORT

Only a vent system was used for cockpit cooling. The cockpit temperature rose as high as 35 degrees Fahrenheit above ambient and hampered the pilots' effectiveness after a short time.

B 108. "Cool suits" should be provided when the aircraft are delivered for Category II (page 50).

ASD COMMENT: ASD concurs. Cool suits have been installed.

*D 109. Air-conditioning should be provided for crew effectiveness (pages and 50).

No crew comfort provisions were made for items such as ash-trays, map cases, foot rests, cup holders, etc.

*D 110. The necessary cross-country and crew comfort cockpit provisions specified in AFSCM 80-1 (C.2-2.12. . .) should be provided (page 8).

STATIC ELECTRICITY

There was no permanently attached static ground wire.

D 111. A permanently attached static ground wire should be provided (page 46).

ASD COMMENT: ASD concurs. Ground wires have been installed.

MANUALS

The original Flight and Maintenance Manuals were outdated due to the aircraft changes made during Category I tests. Cargo loading information was not available, but would be required to properly load the aircraft during Category II Operational Suitability tests.

C 112. The Flight and Maintenance Manuals should be updated (page 46).

ASD COMMENT: ASD concurs. Manuals will be revised.

C 113. A limited Cargo Loading Manual should be provided (page 46).

ASD COMMENT: ASD does not concur; however, cargo loading information and data will be provided to AFFTC.

APPENDIX I

DATA ANALYSIS METHODS AND PLOTS

STANDARD ATMOSPHERE

The data in this report were not corrected to standard day conditions except for the hover ceiling data.

TEST SHAFT HORSEPOWER

Test shaft horsepower was determined by use of the following equation:

$$shp = 3496\ (\%\ rpm)\ (\%\ torque)$$

This equation was determined by substitution of 15 600 rpm (100 % rpm) and 1350 ft-lb torque (100 % torque) into the general equation for shaft horsepower.

HOVER CEILING

Standard day hover ceiling data were calculated from a C_P - C_T nondimensional plot as outlined in reference 9.

CLIMB

Test rate of climb was determined by taking slopes of the pressure altitude versus time curve. Climb data were not corrected to standard conditions, but the difference between test day and standard day ambient temperatures is shown in the climb performance plot (figure 4, appendix I).

LEVEL FLIGHT

Level flight data were obtained at constant altitude by stabilizing at each test altitude and several indicated airspeeds.

Test shaft horsepower was corrected for nonstabilized conditions as follows:

$$shp_{corr} = shp_t - \Delta shp$$

Where:

$$\Delta shp = \left[\frac{(dh/dt)(W_t)}{V_t} + \frac{(dV/dt)(W_t)}{32.2}\right] \frac{V_t}{550}$$

Fuel flow was corrected for nonstabilized conditions by assuming the following relationship:

$$W_{f_{corr}} = W_{f_t} \; (shp_{corr}/shp_t)$$

No other corrections were made to the data, which are presented at test weight conditions.

Although these methods for correcting level flight data for nonstabilized conditions are only approximate, they were necessary since engine and propeller characteristics were not fully defined during the evaluation. Since these corrections were small, the errors introduced were assumed to be negligible.

TAKEOFF AND LANDING PERFORMANCE

Fairchild Flight Analyzers were used to record the data through 50 feet. Wind data were collected, but wind corrections were not applied since all tests were conducted under crosswind conditions and on reciprocal headings. No other corrections were made.

TABLE XVII

COOPER RATINGS

	Adjective rating	Numerical rating	Description	Primary mission accomplished	Can be landed
Normal operation	Satisfactory	1	Excellent, includes optimum	yes	yes
		2	Good, pleasant to fly	yes	yes
		3	Satisfactory, but with some mildly unpleasant characteristics	yes	yes
Emergency operation	Unsatisfactory	4	Acceptable, but with unpleasant characteristics	yes	yes
		5	Unacceptable for normal operation	doubtful	yes
		6	Acceptable for emergency condition only	doubtful	yes
No operation	Unacceptable	7	Unacceptable even for emergency condition	no	doubtful
		8	Unacceptable - dangerous	no	no
		9	Unacceptable - uncontrollable	no	no
No operation	Catastrophic	10	Motions possibly violent enough to prevent pilot escape	no	no

XC-142A USAF S/N 62-5923
T64-GE-1 Engines 2FE16A3-4A Propellers
Four Engines Four Propellers Operating
Maximum Rated Power 95% Propeller rpm
$C_P - C_T$ Method
Standard Day Conditions

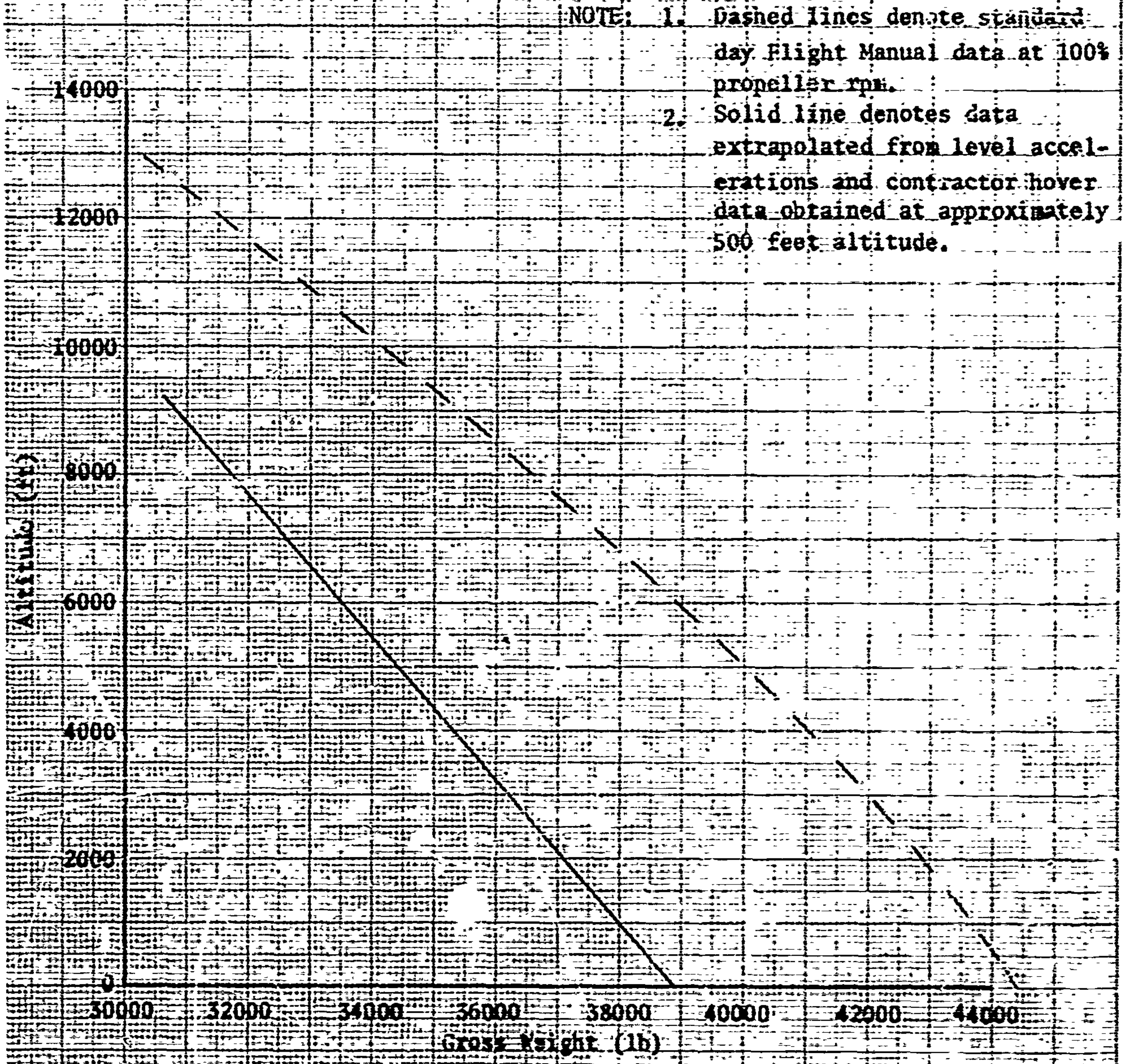


Figure 1. Hover Performance

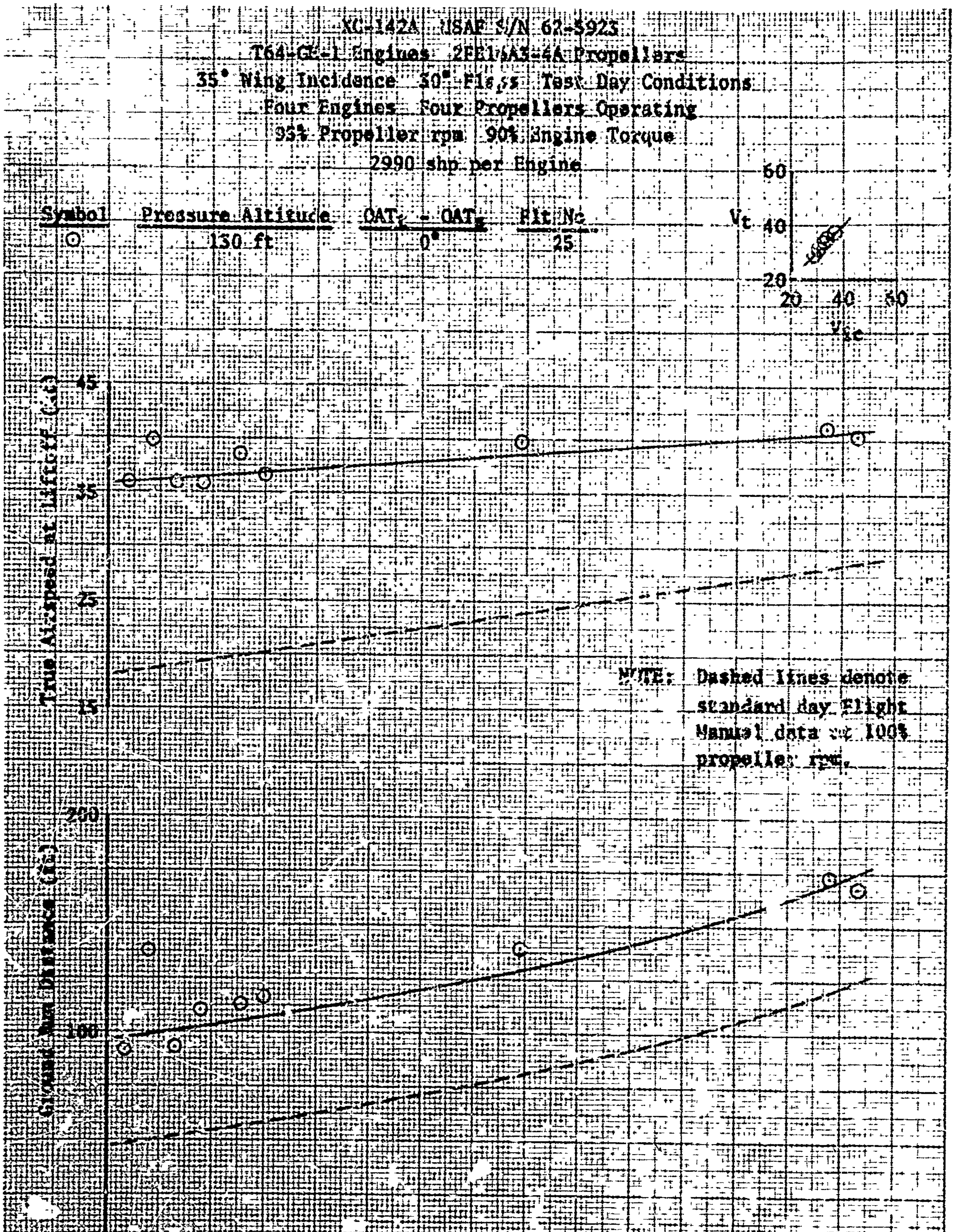


Figure 2. Takeoff Performance

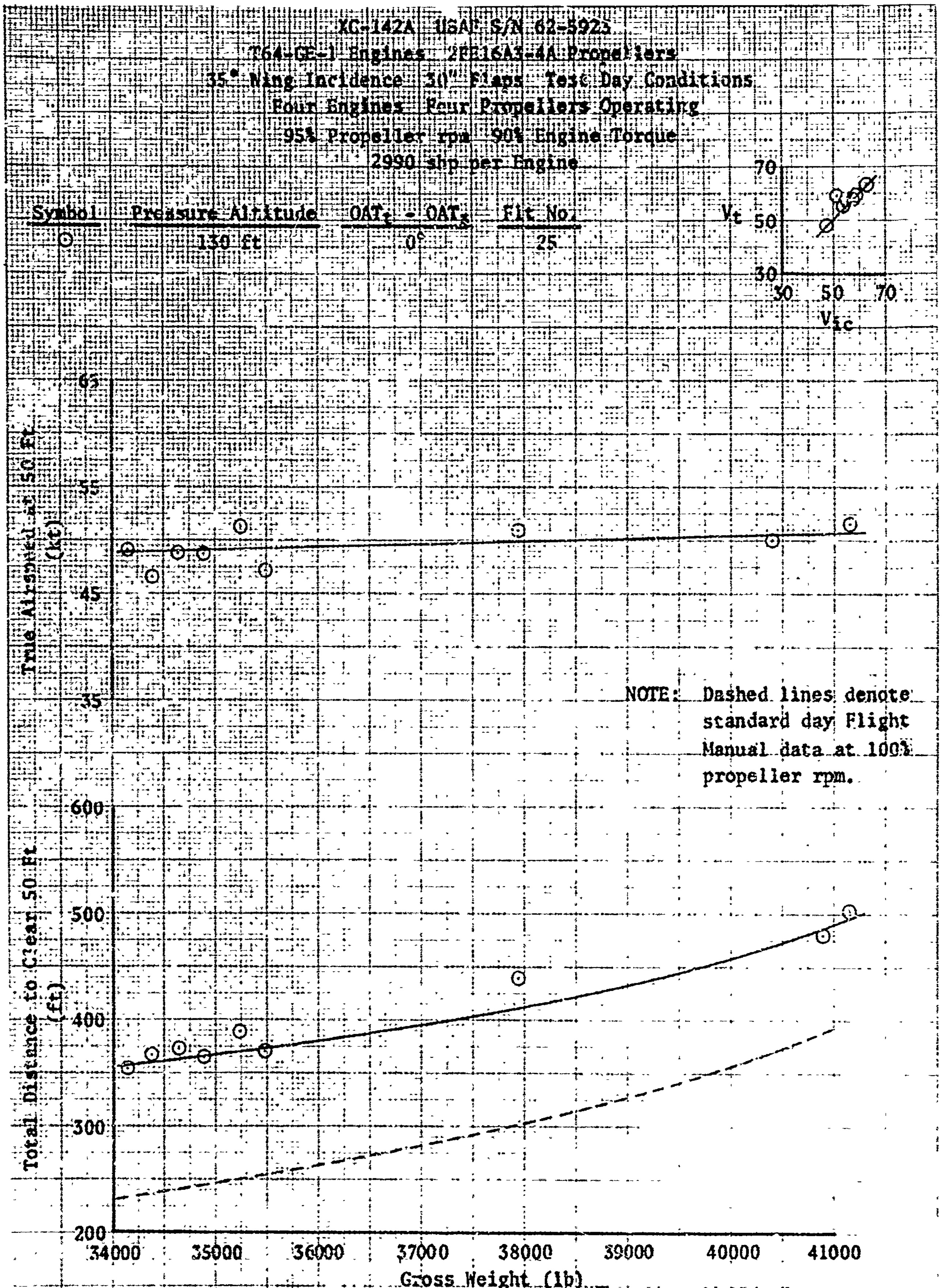


Figure 3. Takeoff Over 50 Ft Obstacle

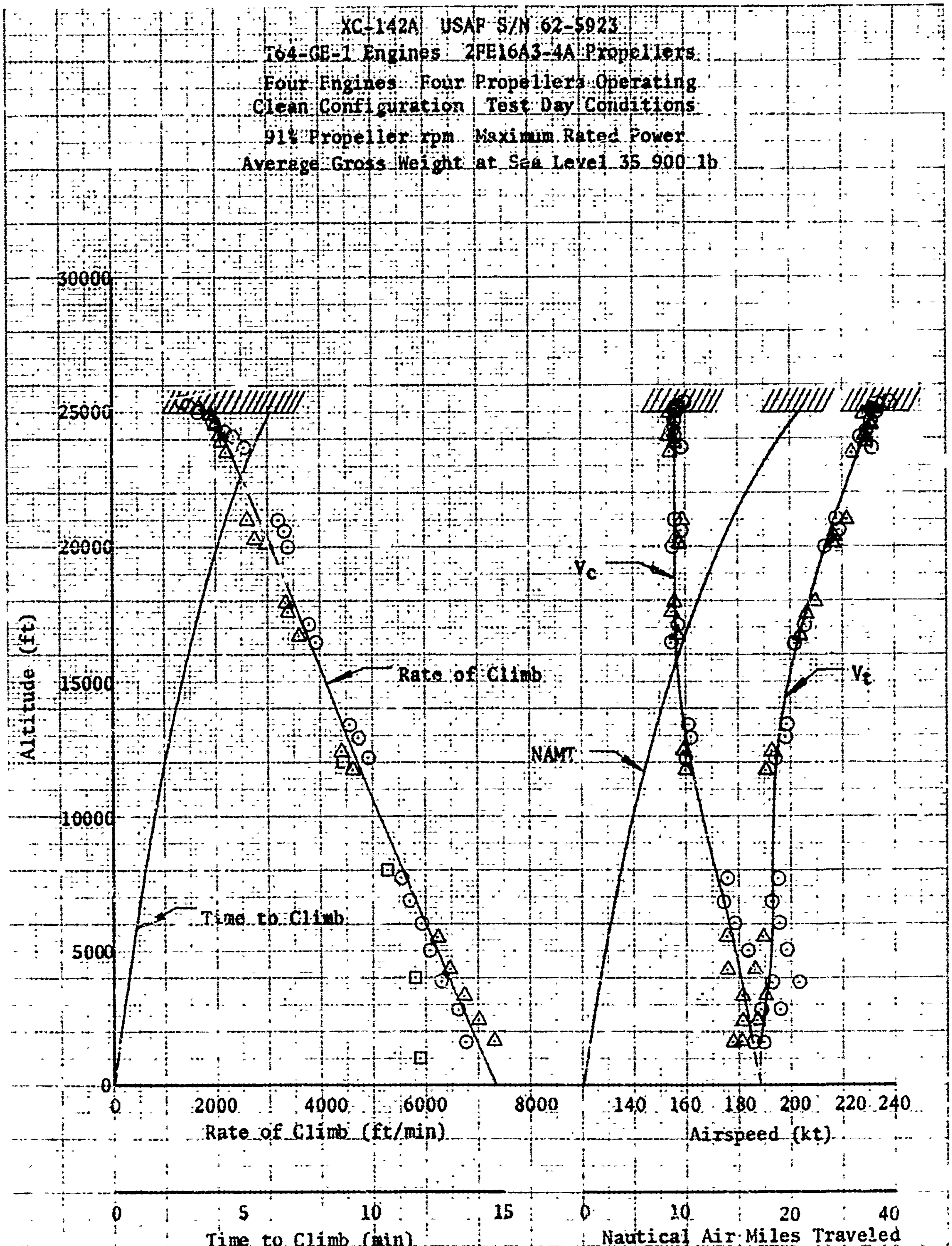


Figure 4. Climb Performance

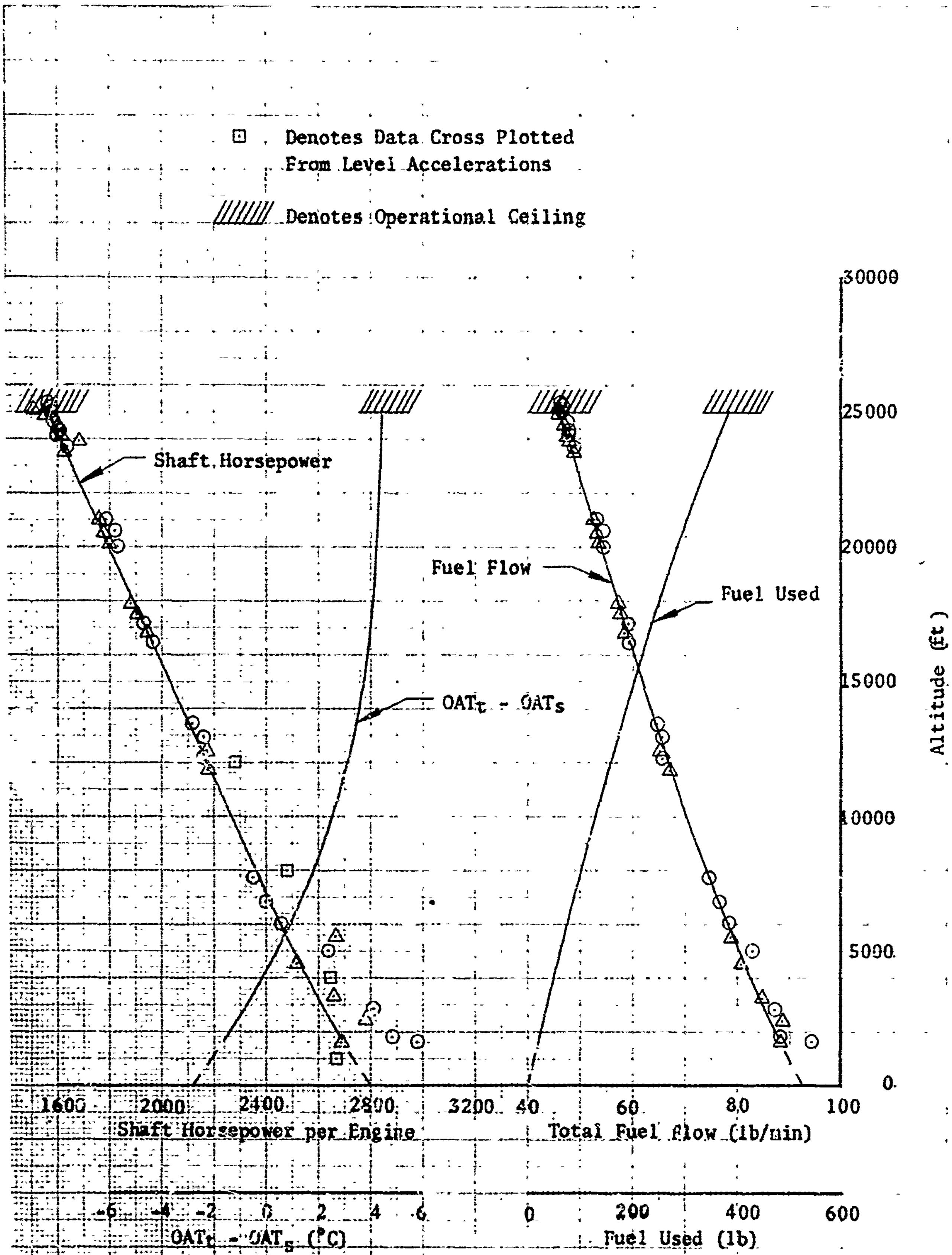


Figure 4. Climb Performance (concluded)

XC-142A USAF S/N 62-5923
T64-GE-1 Engines 2FE16A3-4A Propellers
Four Engines Four Propellers Operating
Clean Configuration Test Day Conditions
95% Propeller rpm Maximum Rated Power

Symbol	Altitude (ft)	Gross Weight (lb)	$OAT_t - OAT_s$ (°C)
⊙	1 000	34 500	+13
⊡	4 000	34 950	+13
◇	8 000	35 280	+14
△	12 000	35 550	+16

NOTE: Tails Denote Points Flown In Opposite Direction.

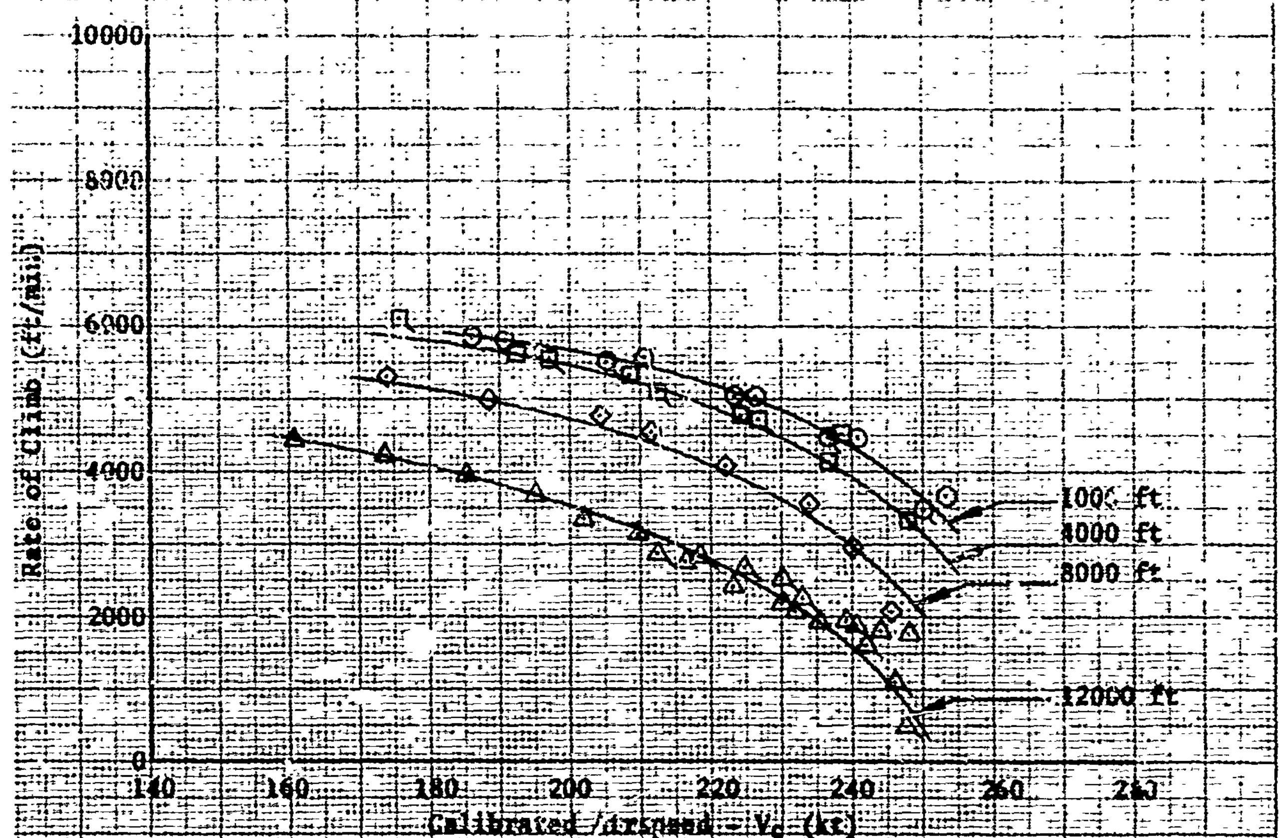


Figure 5. Climb Potential

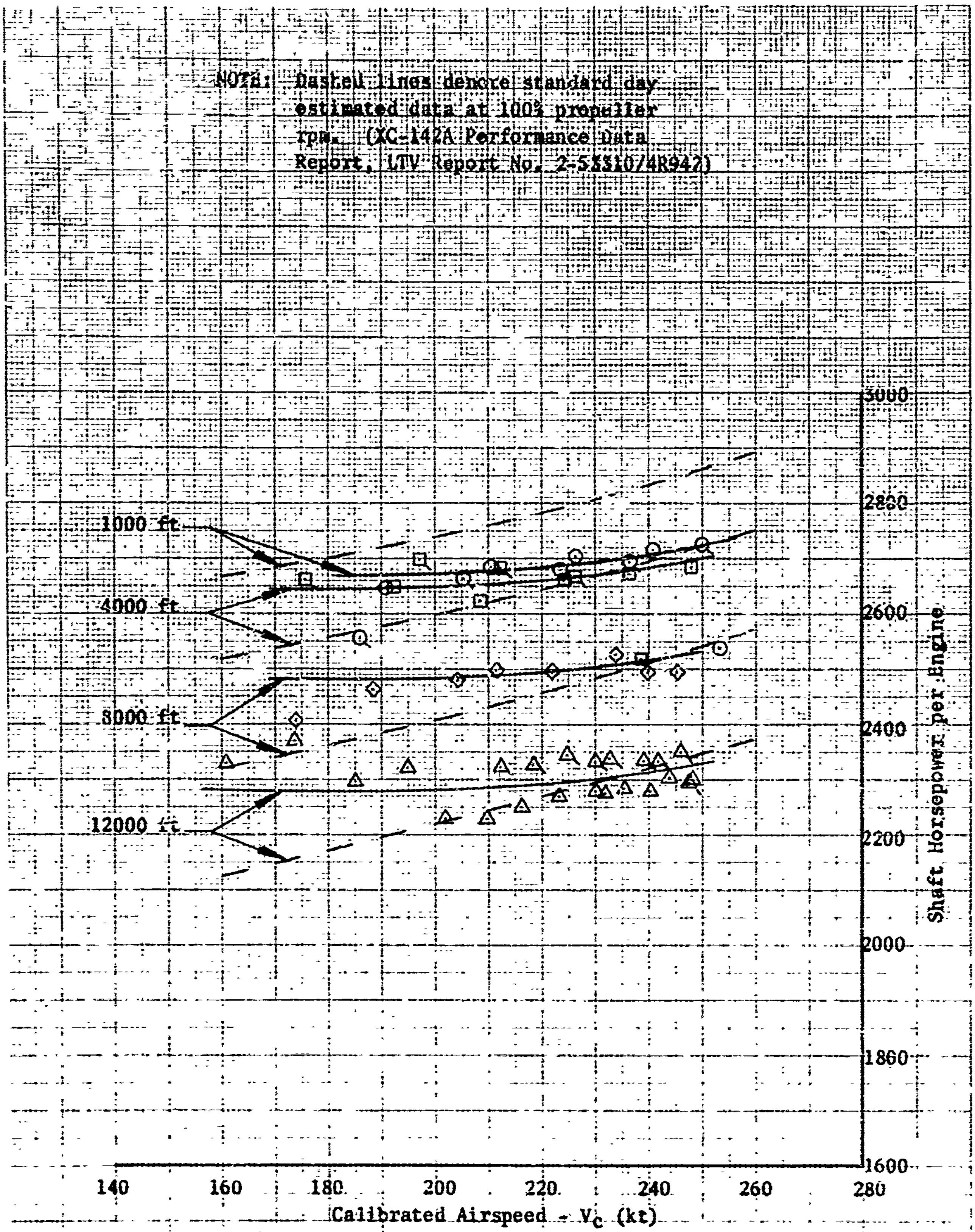


Figure 5. Climb Potential (concluded)

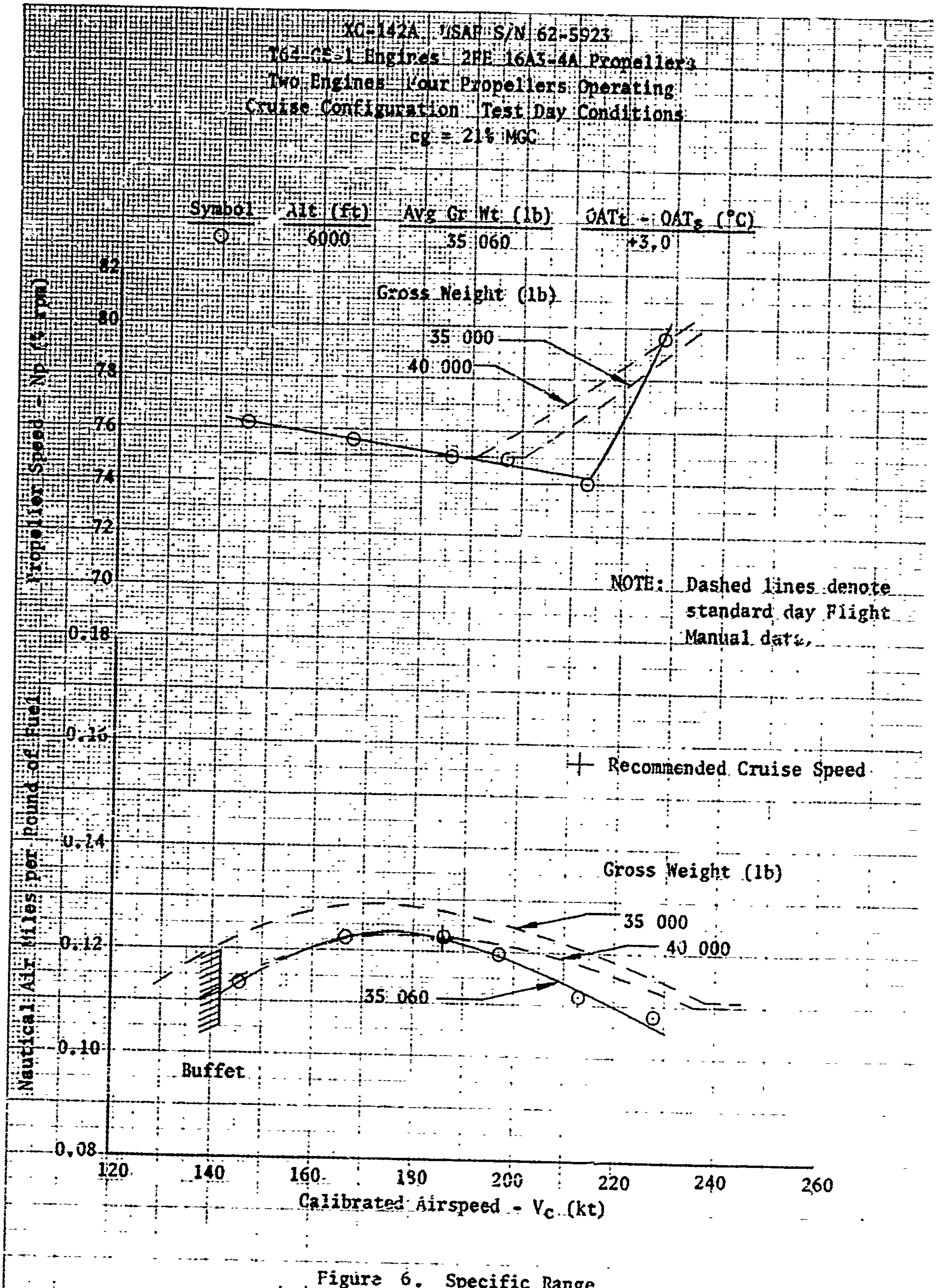


Figure 6. Specific Range

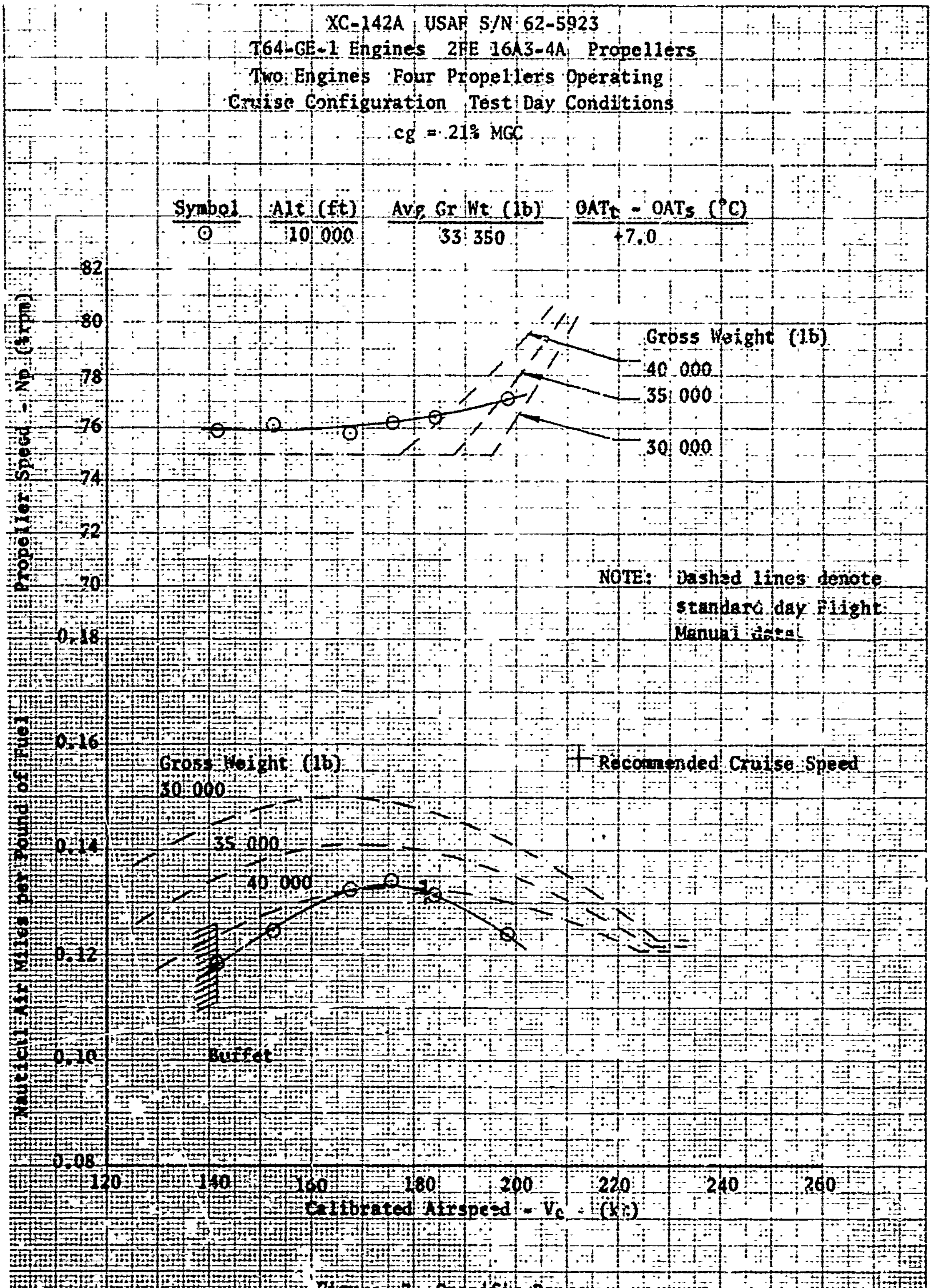


Figure 7. Specific Range

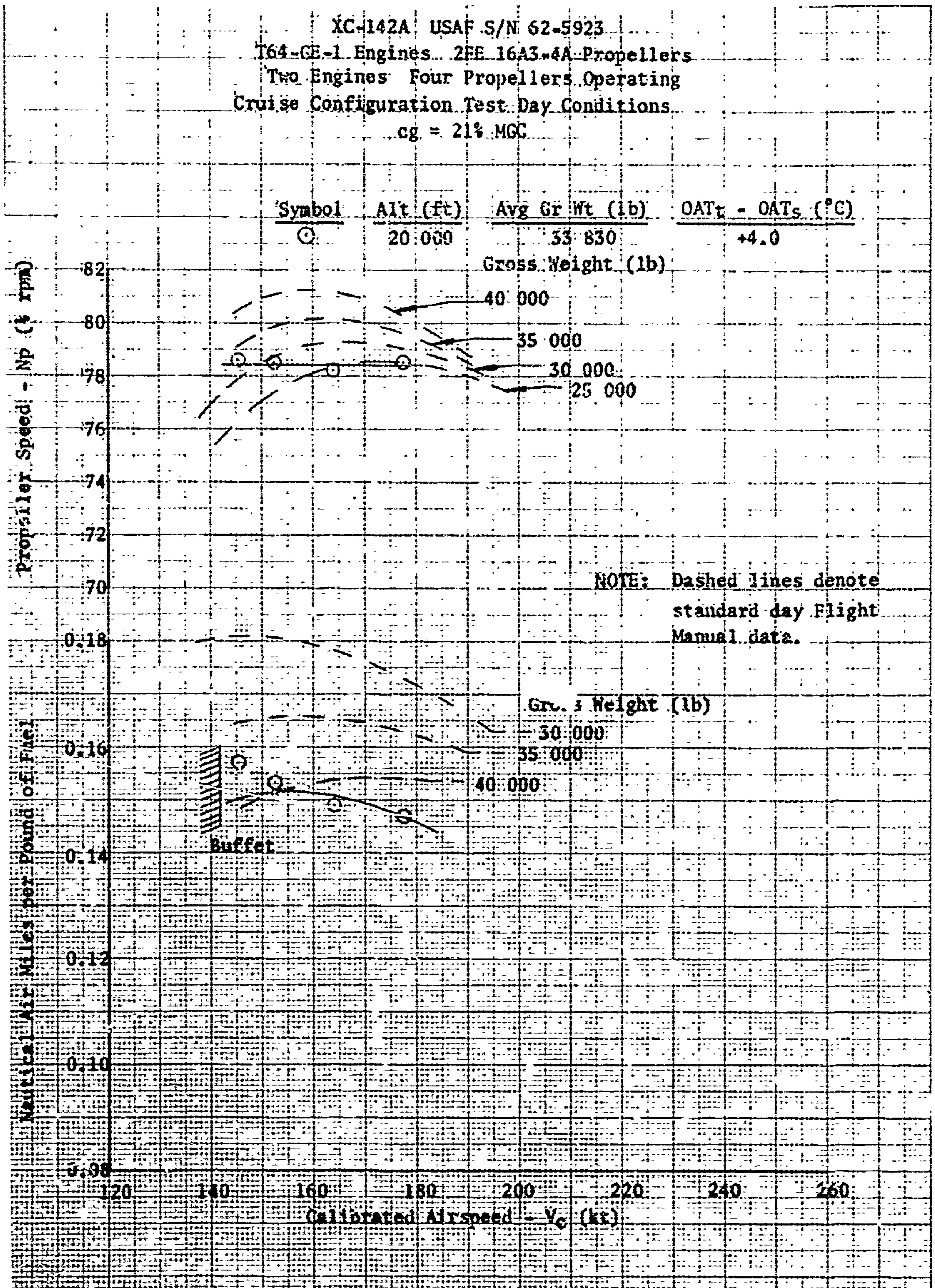


Figure 8. Specific Range

XC-142A USAF S/N 62-5923
T64-GE-1 Engines 2FE16A3-4A Propellers
Four Engines, Four Propellers Operating
Cruise Configuration - Test Day Conditions
cg = 21% MGC

Sym	Alt (ft)	Avg G.W. (lb)	OAT_t-OAT_s (°C)
⊙	10 000	34 750	+8.0
⊡	25 000	35 510	+2.0

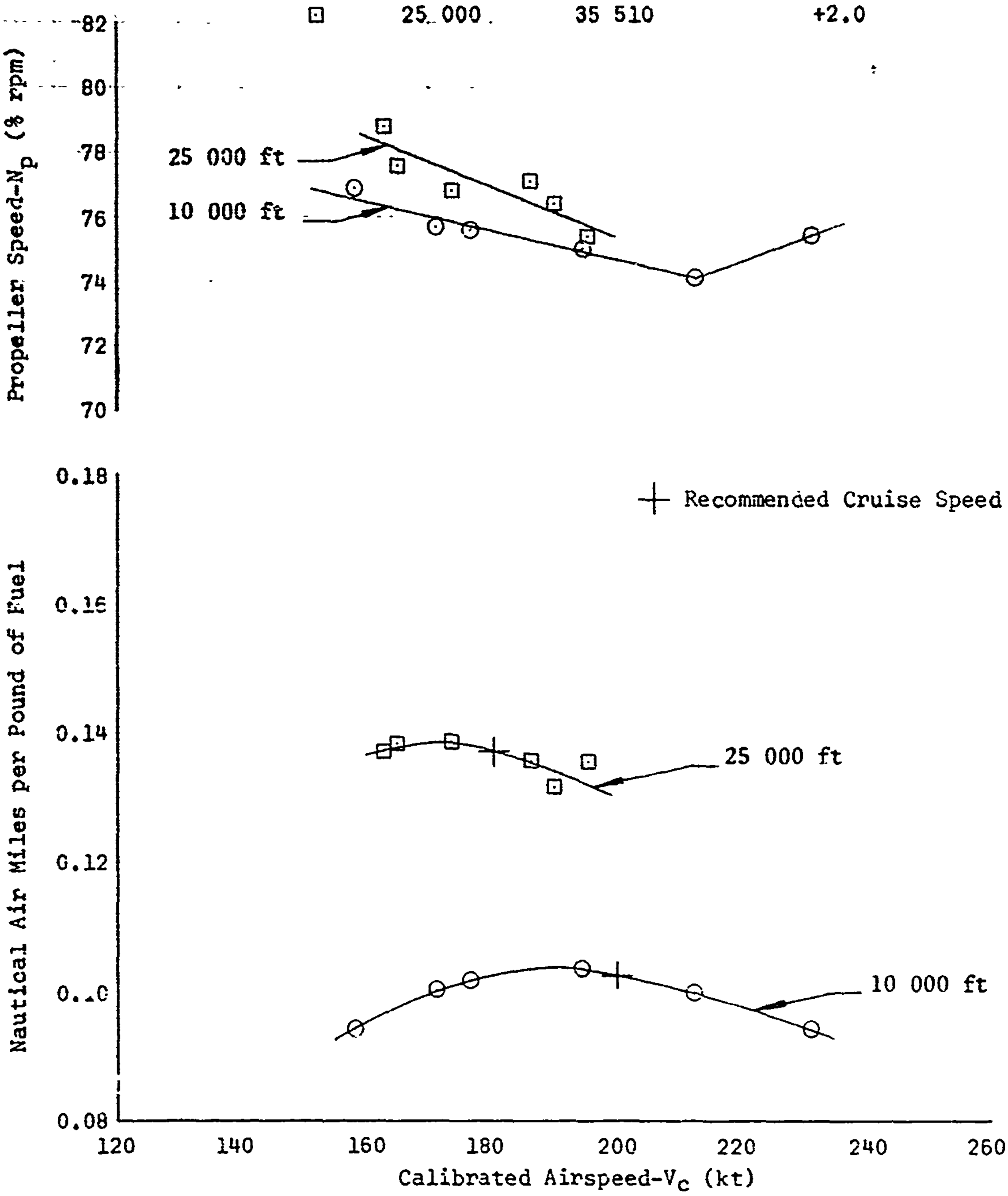


Figure 9. Specific Range

XC-142A USAF S/N 62-5923
T64-GE-1 Engines 2FE16A3-4A Propellers
Cruise Configuration Test Day Conditions
Four Propellers Operating
W_{iw} - 35 000 lb
cg = 21% MGC

Symbol	Altitude (ft)	Gross Weight (lb)	No. Engines
⊙	6 000	35 060	2
⊡	10 000	33 350	2
⊡ (flagged)	10 000	34 750	4
◇	20 000	35 830	2
△	25 000	35 510	4

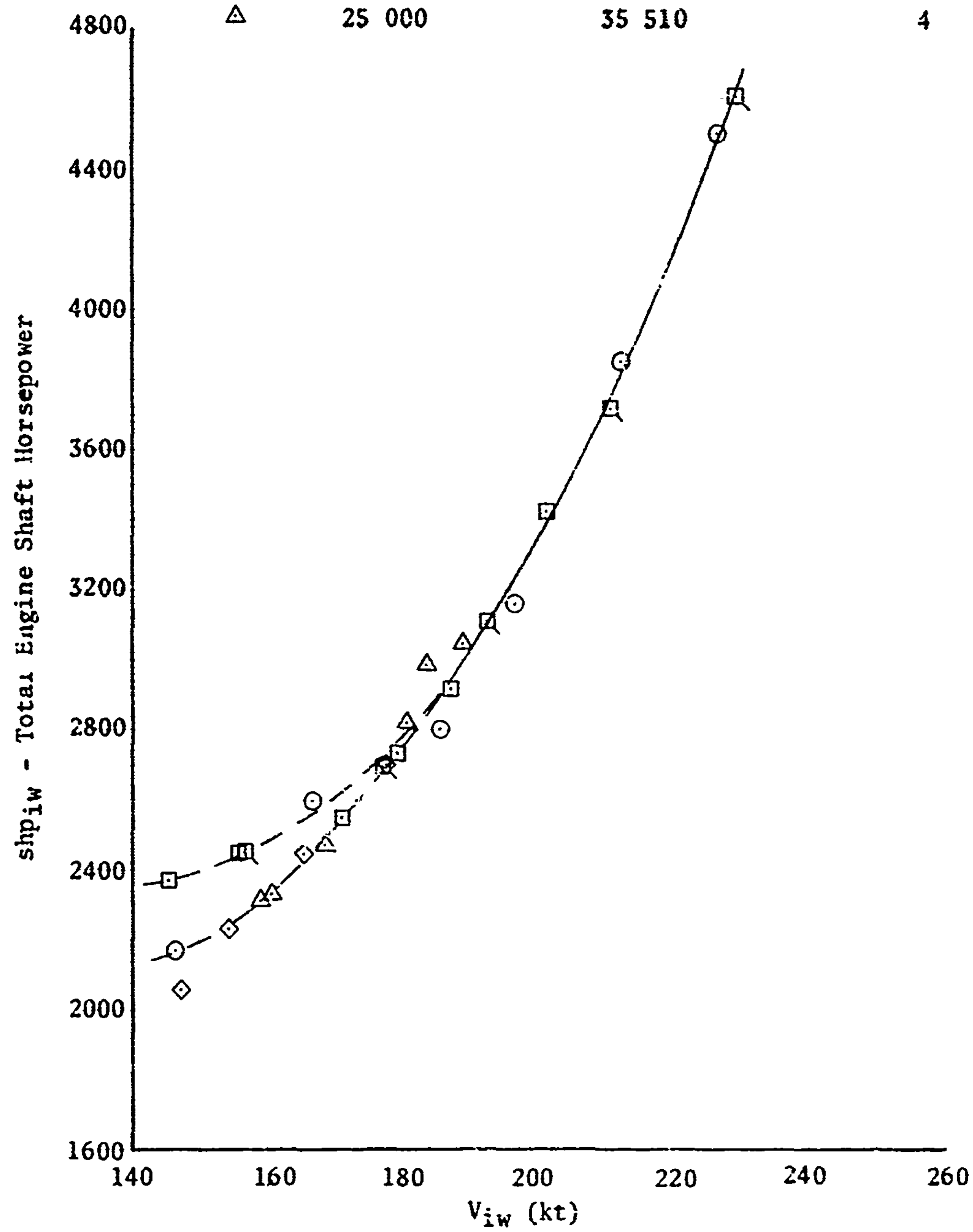


Figure 10. shp_{iw} vs V_{iw}

XC-142A USAF S/N 62-5923
T64-GE-1 Engines 2FE16A3-4A Propellers
35° Wing Incidence 60° Flaps Test Day Conditions
Four Engines Four Propellers Operating
95% Propeller rpm

Symbol	Pressure Altitude	$OAT_t - OAT_s$	Flt No.
⊙	130 ft	0°	25

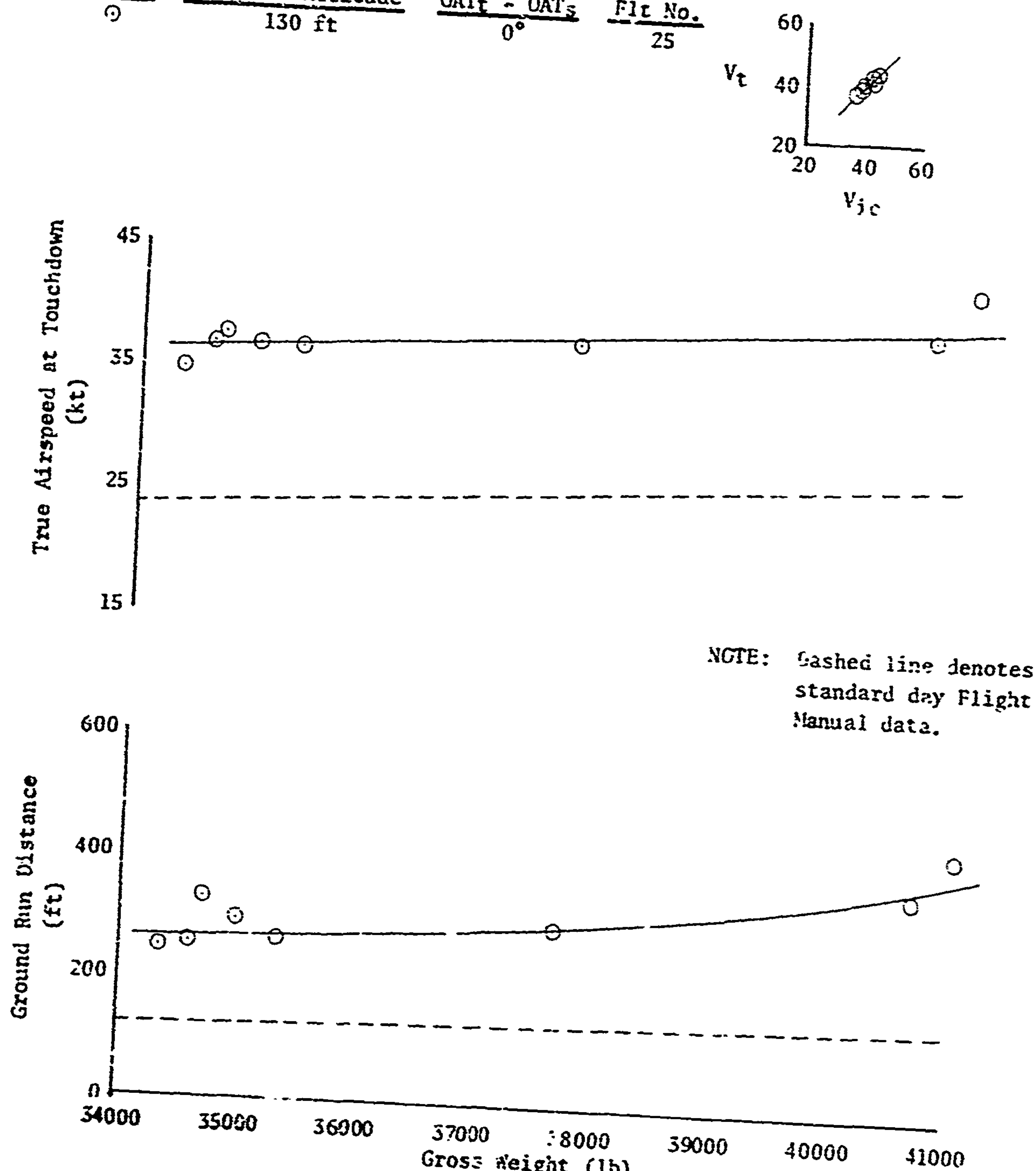


Figure 11. Landing Performance

XC-142A USAF S/N 62-5923
T64-GE-1 Engines 2FE16A2-4A Propellers
35° Wing Incidence 60° Flaps Test Day Conditions
Four Engines Four Propellers Operating
95% Propeller rpm

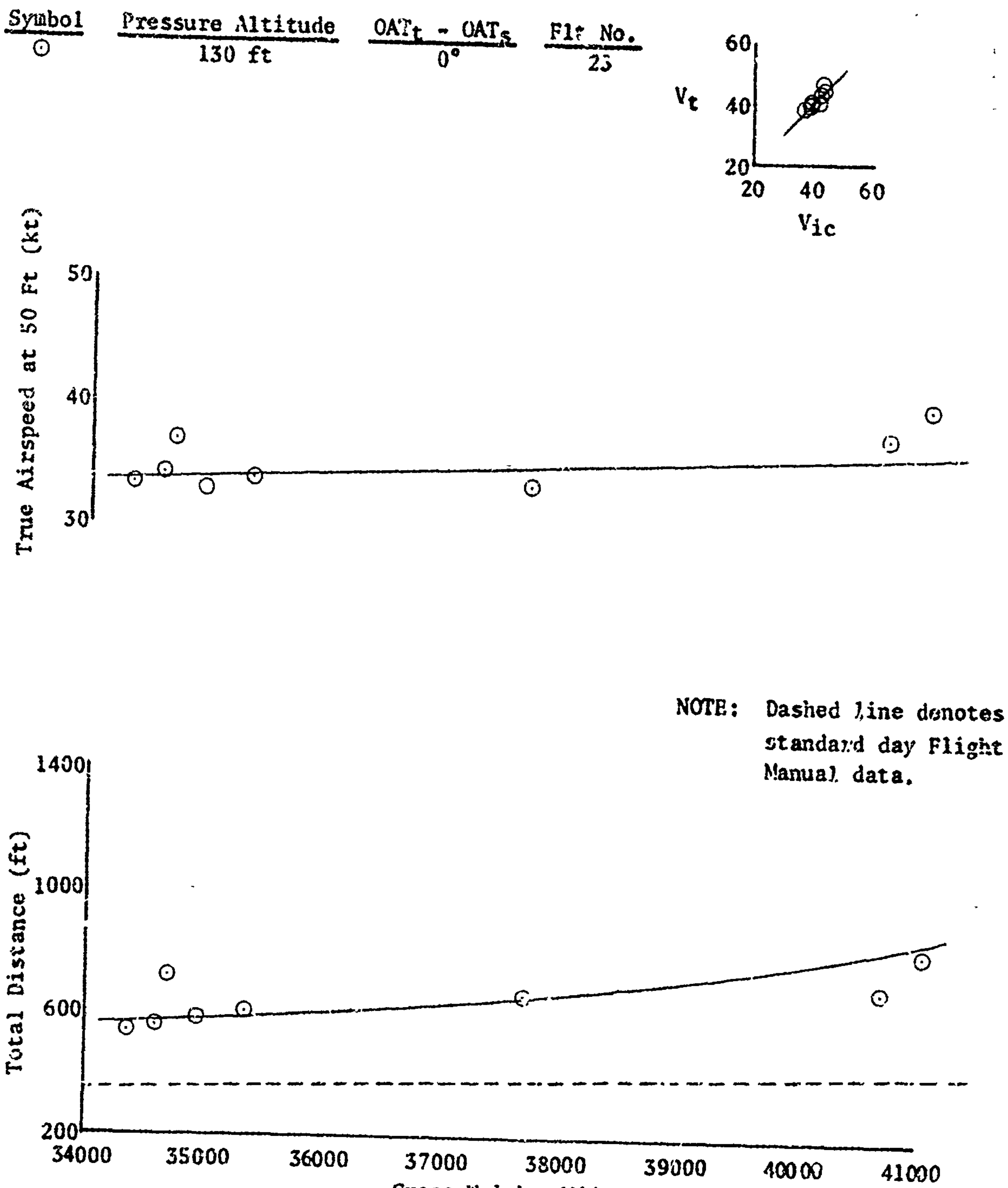


Figure 12. Landing Over 50 Ft Obstacle

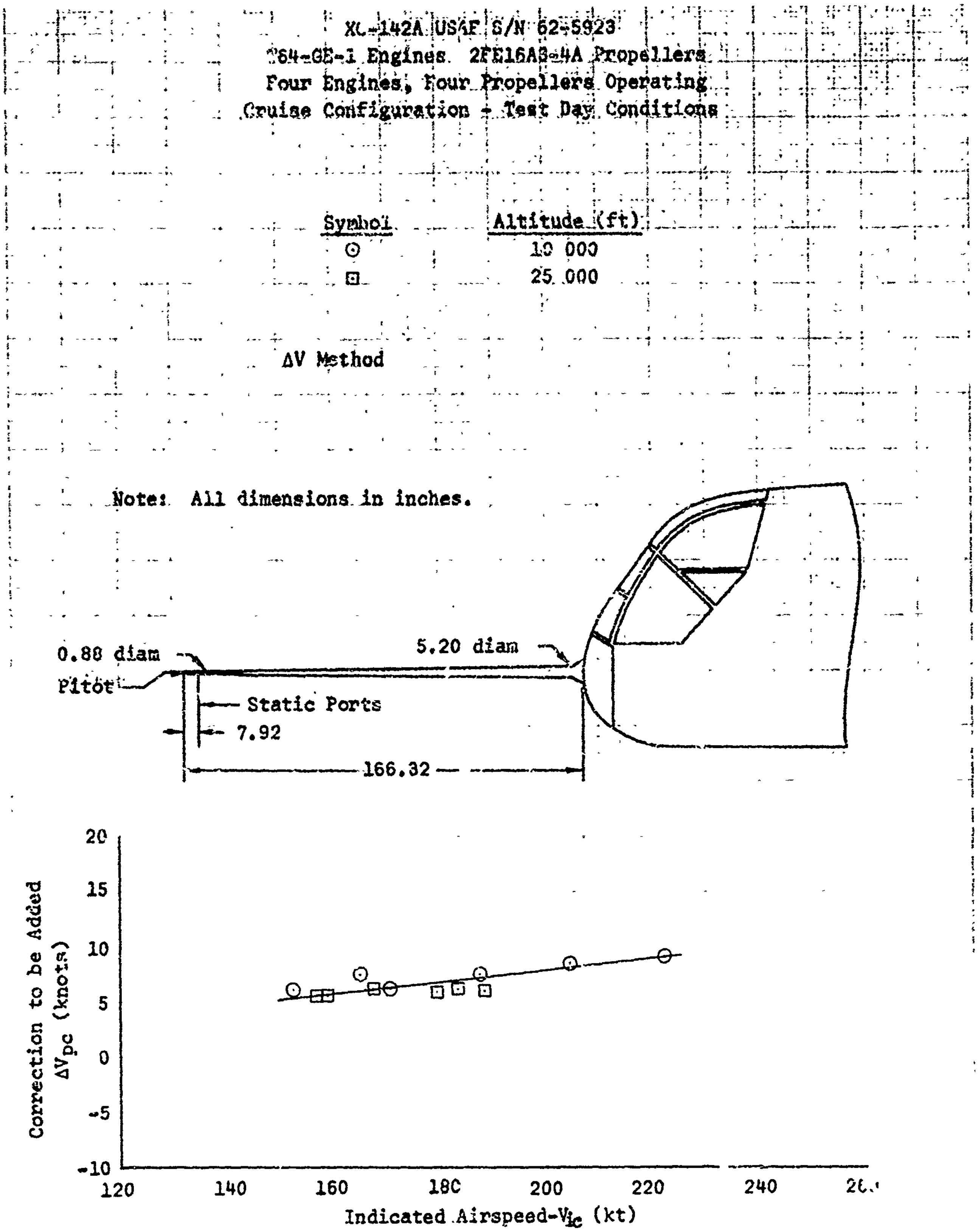


Figure 13 . Airspeed Calibration

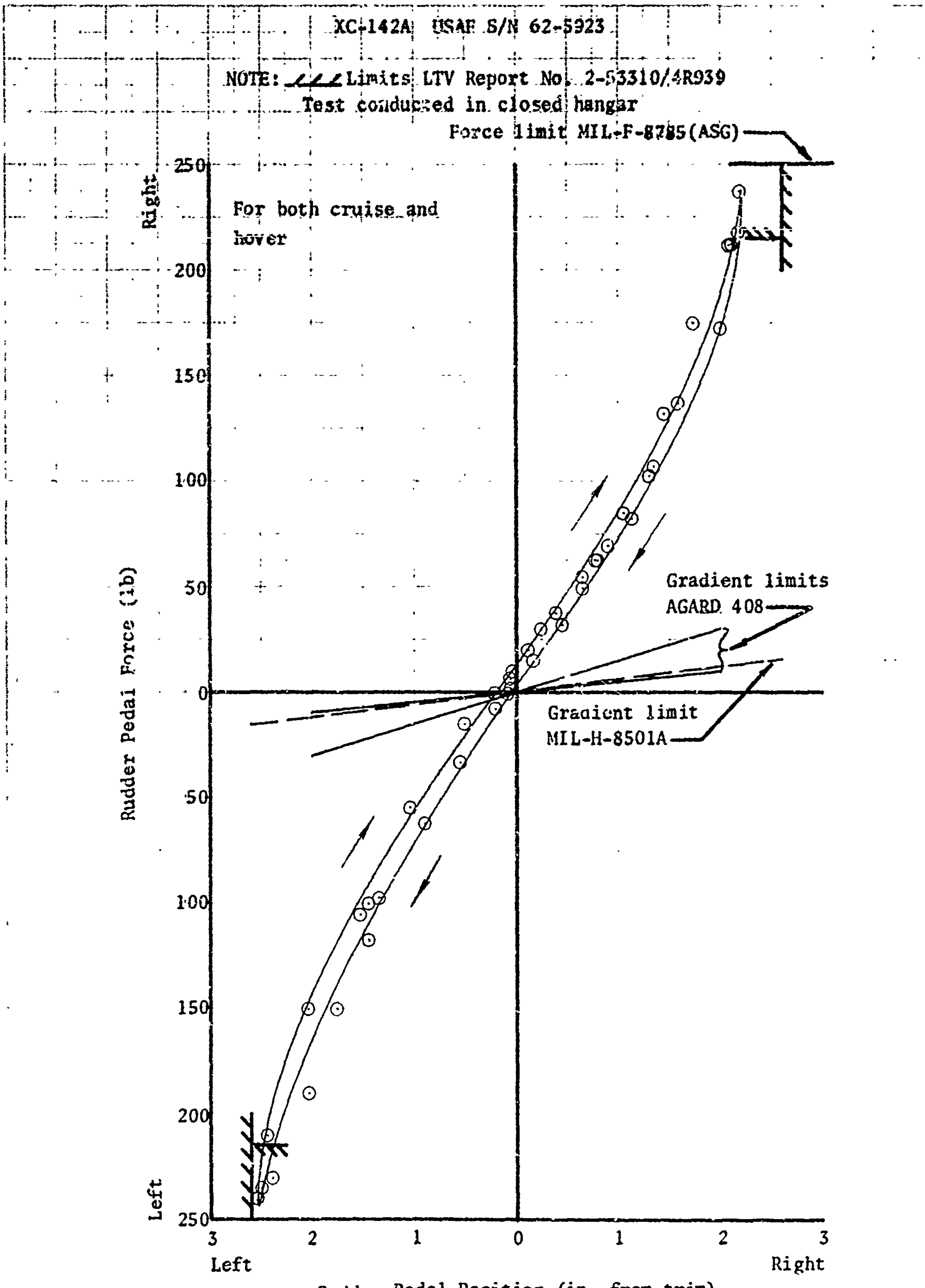


Figure 14. Rudder Pedal Position vs Force

XC-142A USAF S/N 62-5923
Hover Configuration

Symbol	Flight No.	Avg Gr Wt (lb)	Avg cg (% MGC)	Alt (ft)
○	32	32 920	19.2	470
□	28	34 280	19.1	500

▲ Limits AGARD 408 ◆ Limits MIL-H-8501A

NOTE: ⫽⫽⫽ Limits LTV Report No. 2-53310/4R939
Pitch, Roll, and Yaw SAS: ON
Altitude Damper: OFF
4 Knot Wind Favoring Left Yaw

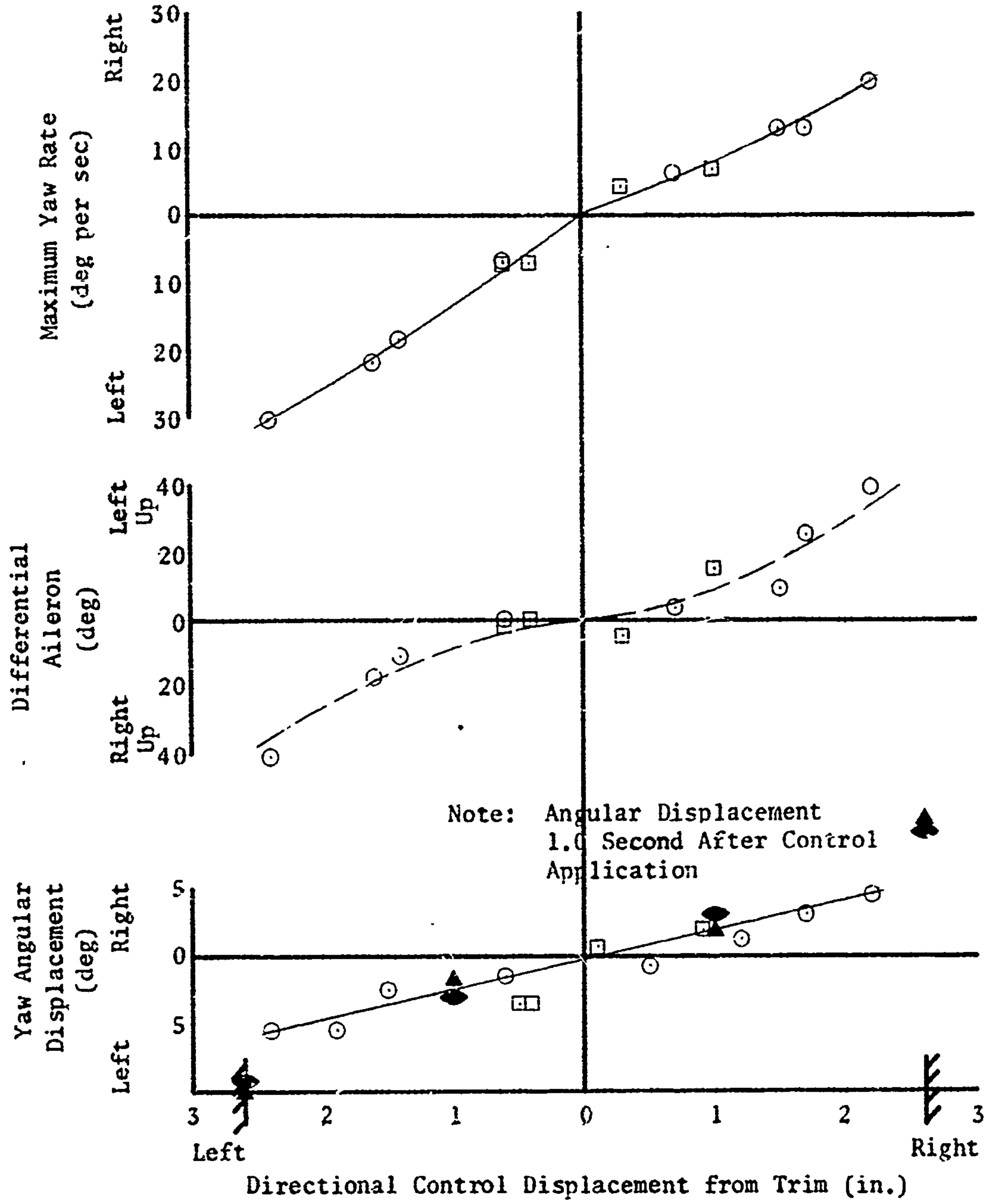


Figure 15. Directional Controllability

XC-142A USAF S/N 62-5923
Hover Configuration

Symbol	Flight No.	Avg Gr Wt (lb)	Avg cg (% MGC)	Alt (ft)
○	33	33 930	19.1	450
□	32	34 030	19.1	450

NOTE: Pitch, Roll, and Yaw SAS: ON
Altitude Damper: OFF

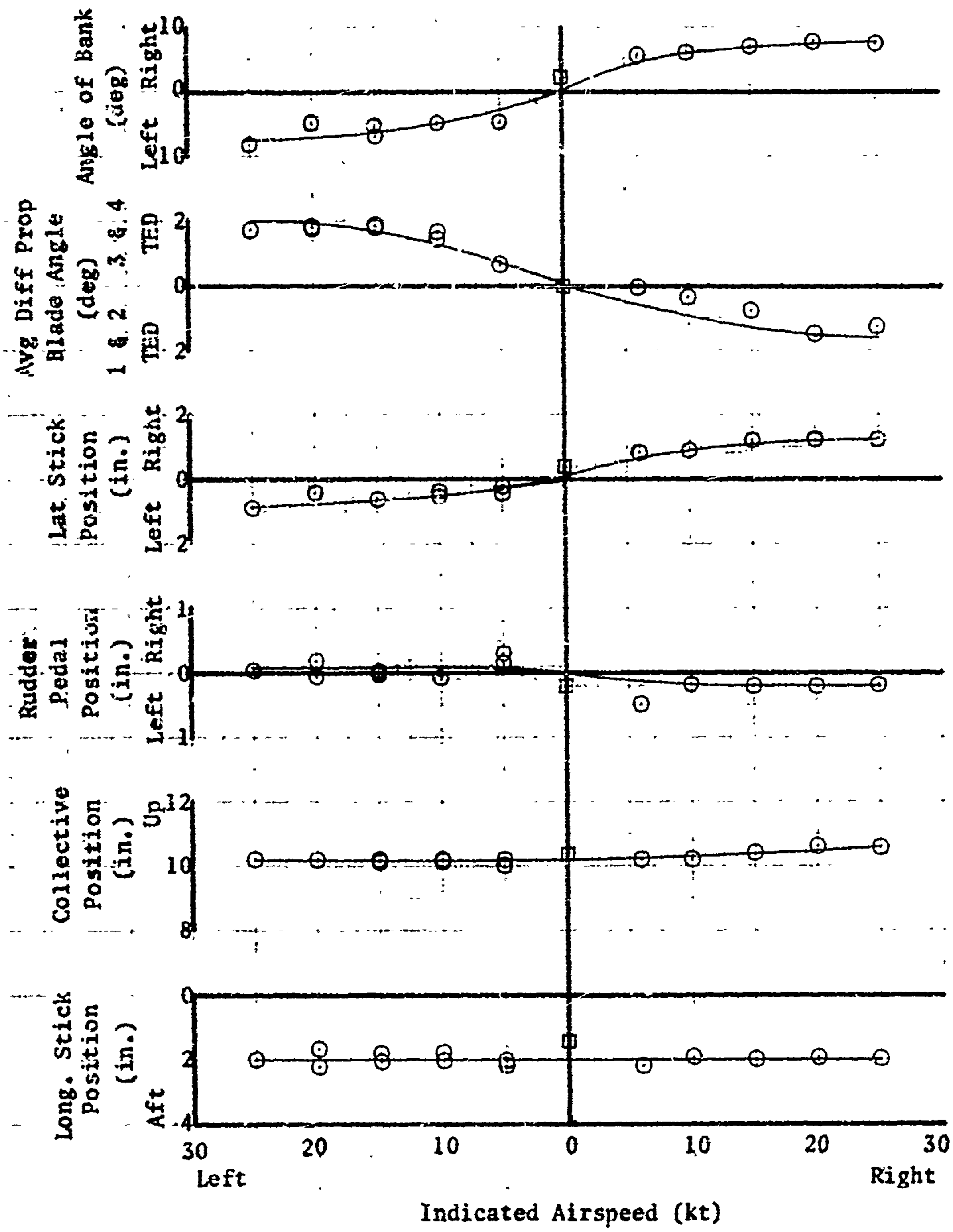


Figure 16. Control Positions in Sideward Flight

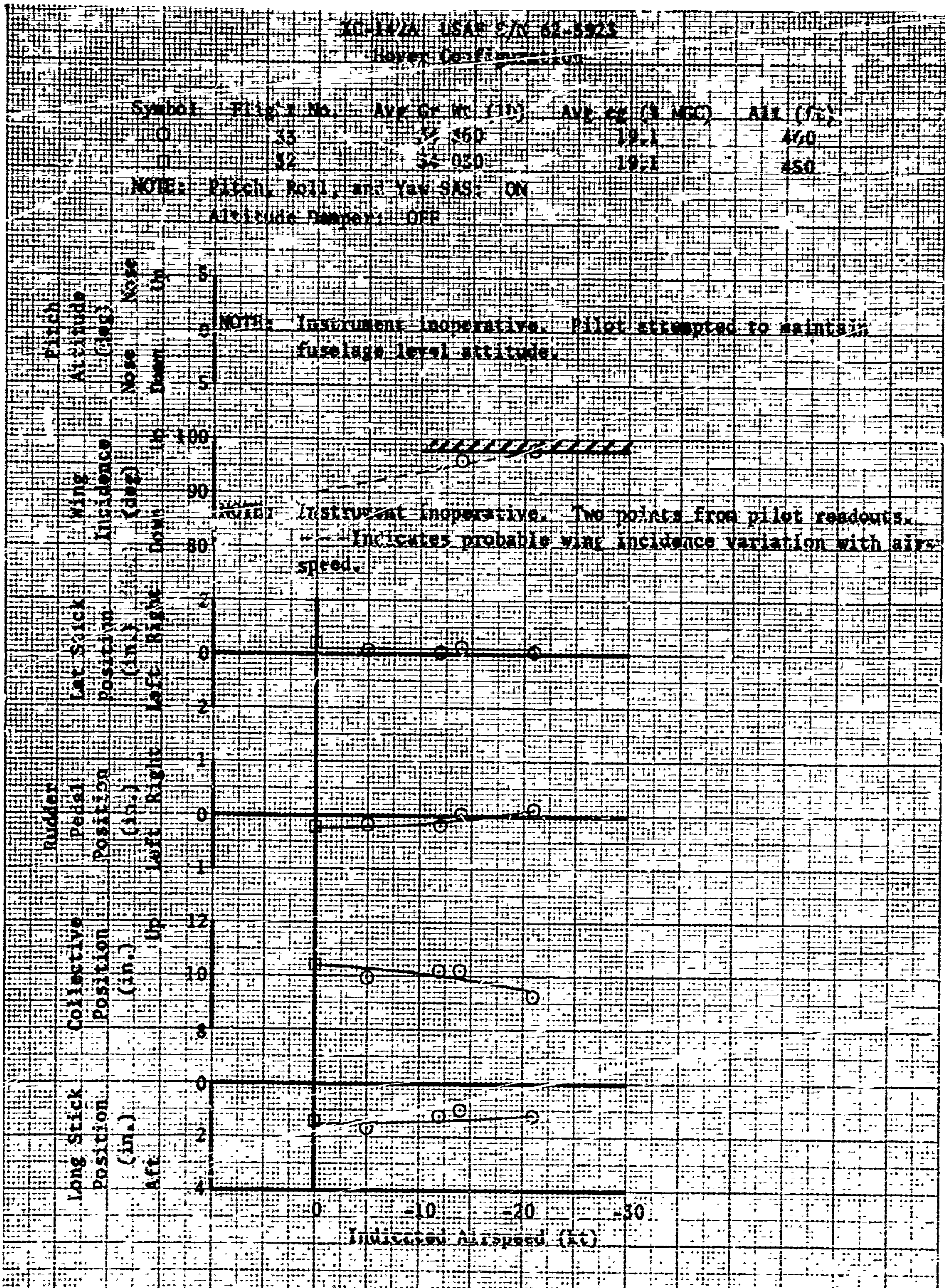


Figure 17. Control Positions in Rearward Flight

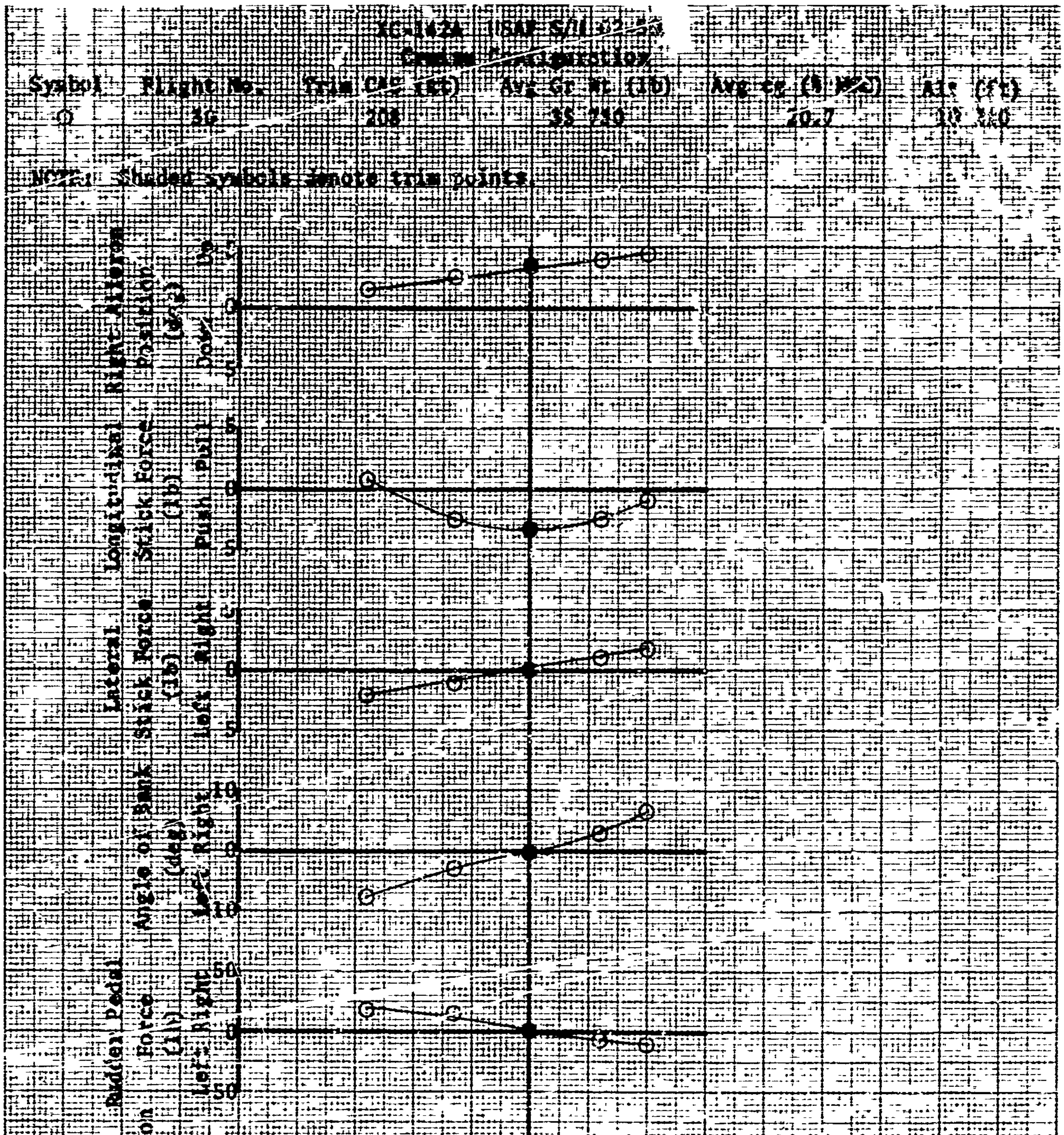


Figure 18. Static Directional Stability

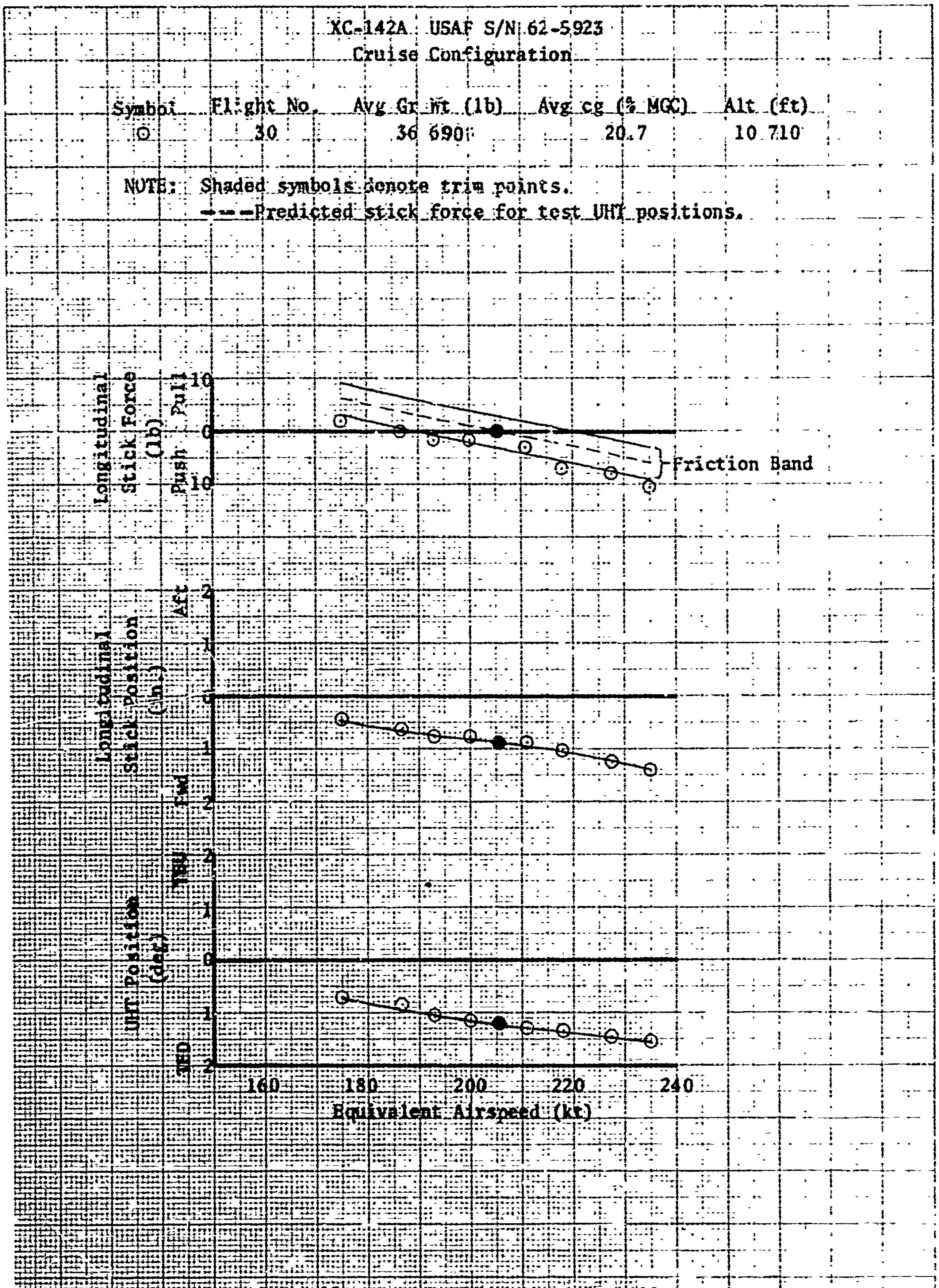


Figure 19. Static Longitudinal Stability

XC-142A USAF S/N 62-5923

Cruise Configuration

Symbol	Turn	Flight No.	Trim CAS (kt)	Avg Gr Wt (lb)	Avg cg (% MAC)	Alt (ft)
○	Left	30	210	35 880	20.7	11 250
□	Right	30	210	35 980	20.7	11 040

NOTE: ---Limits LTV Report No. 2-53310/4R939 [Class III airplane]

////Limits MIL-F-8785(ASG) [Class II airplane]

Shaded symbols denote trim points

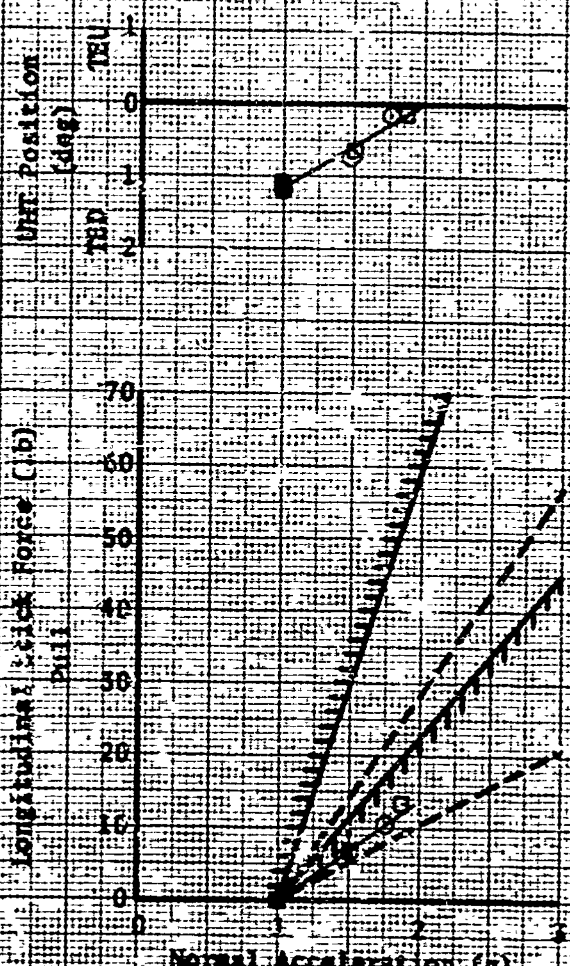
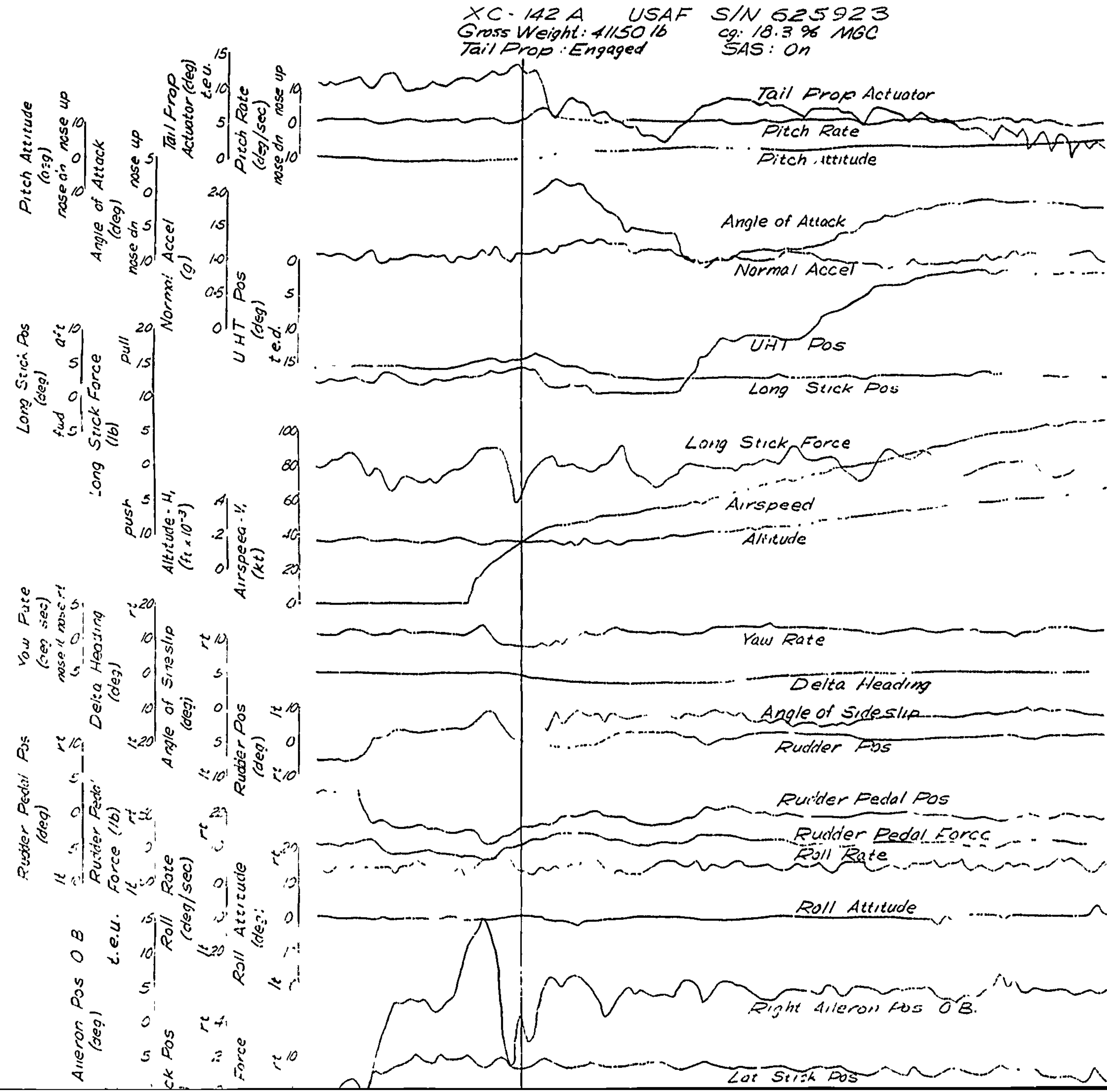


Figure 20. Maneuvering Flight


XC-142 A USAF S/N 625923
Gross Weight: 41150 lb
cg: 18.3 % MGC
Tail Prop: Engaged
SAS: On
Tail Prop Actuator
Pitch Rate
Pitch Attitude
Angle of Attack
Normal Accel
UHT Pos
Long Stick Pos
Long Stick Force
Airspeed
Altitude
Yaw Rate
Delta Heading
Angle of Sideslip
Rudder Pos
Rudder Pedal Pos
Rudder Pedal Force
Roll Rate
Roll Attitude
Right Aileron Pos O.B.
Lat Stick Pos
Pitch Attitude (deg)
Angle of Attack (deg)
Tail Prop Actuator (deg)
Pitch Rate (deg/sec)
Long Stick Pos (deg)
Long Stick Force (lb)
Normal Accel (g)
UHT Pos (deg)
Altitude - H, (ft x 10^-3)
Airspeed - V, (kt)
Yaw Rate (deg/sec)
Delta Heading (deg)
Angle of Sideslip (deg)
Rudder Pos (deg)
Rudder Pedal Pos (deg)
Rudder Pedal Force (lb)
Roll Rate (deg/sec)
Roll Attitude (deg)
Aileron Pos O B (deg)

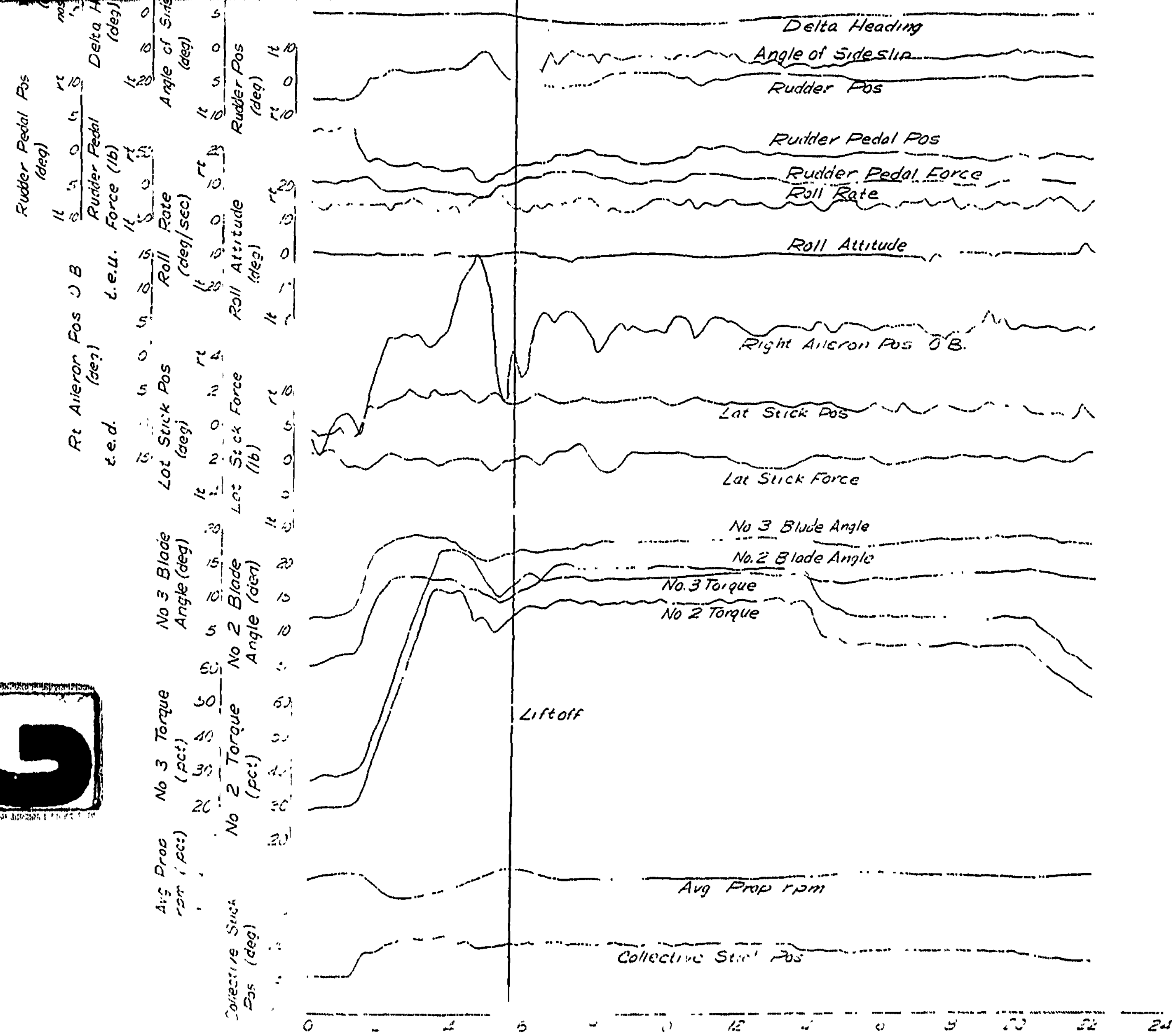
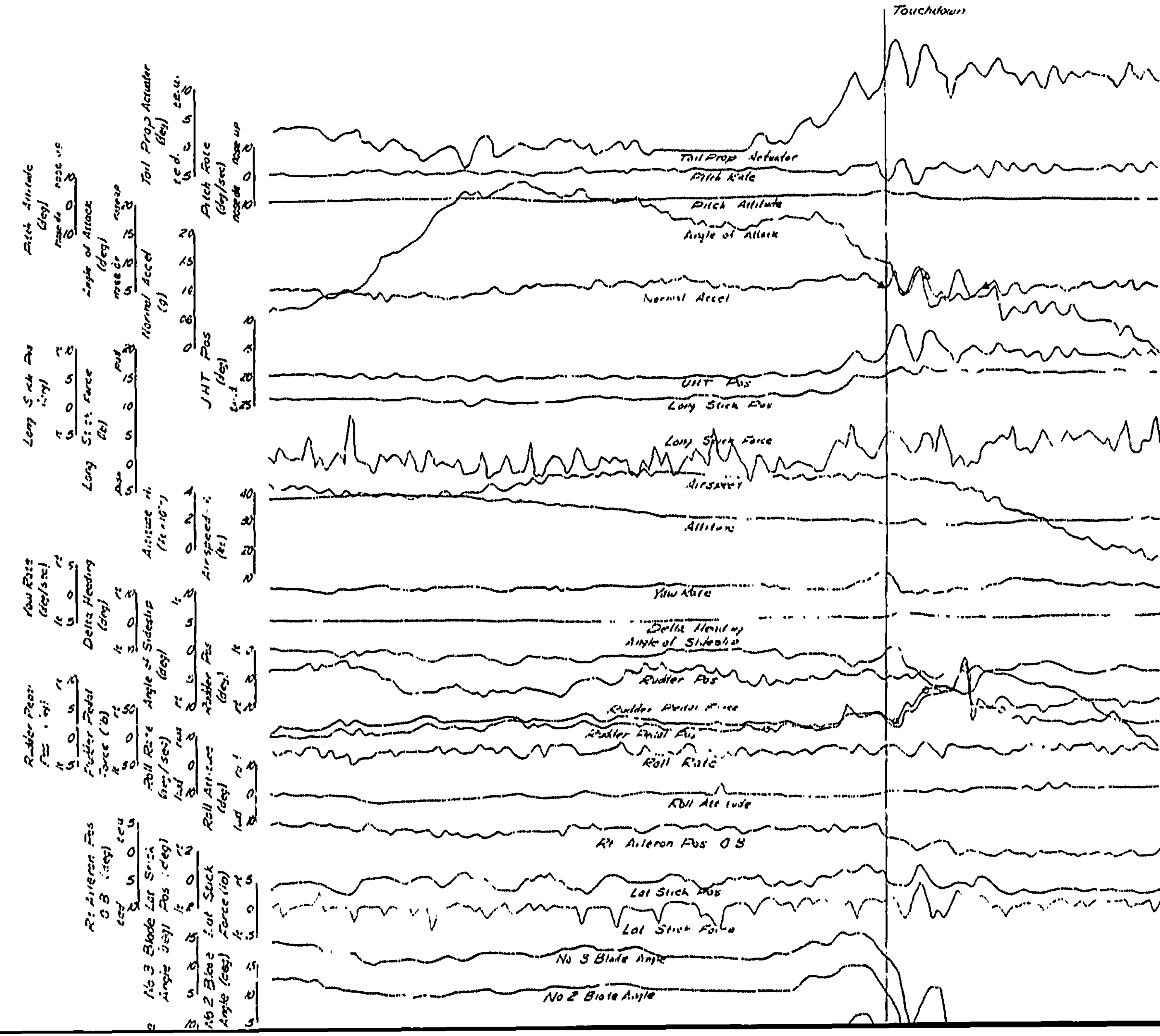


Figure 21 35°, 30' Takeoff

XC-142 A USAF S/N 62-5923
Gross Weight : 41050 lb cg : 18.3 MGC
Tail Prop Engaged SAS On



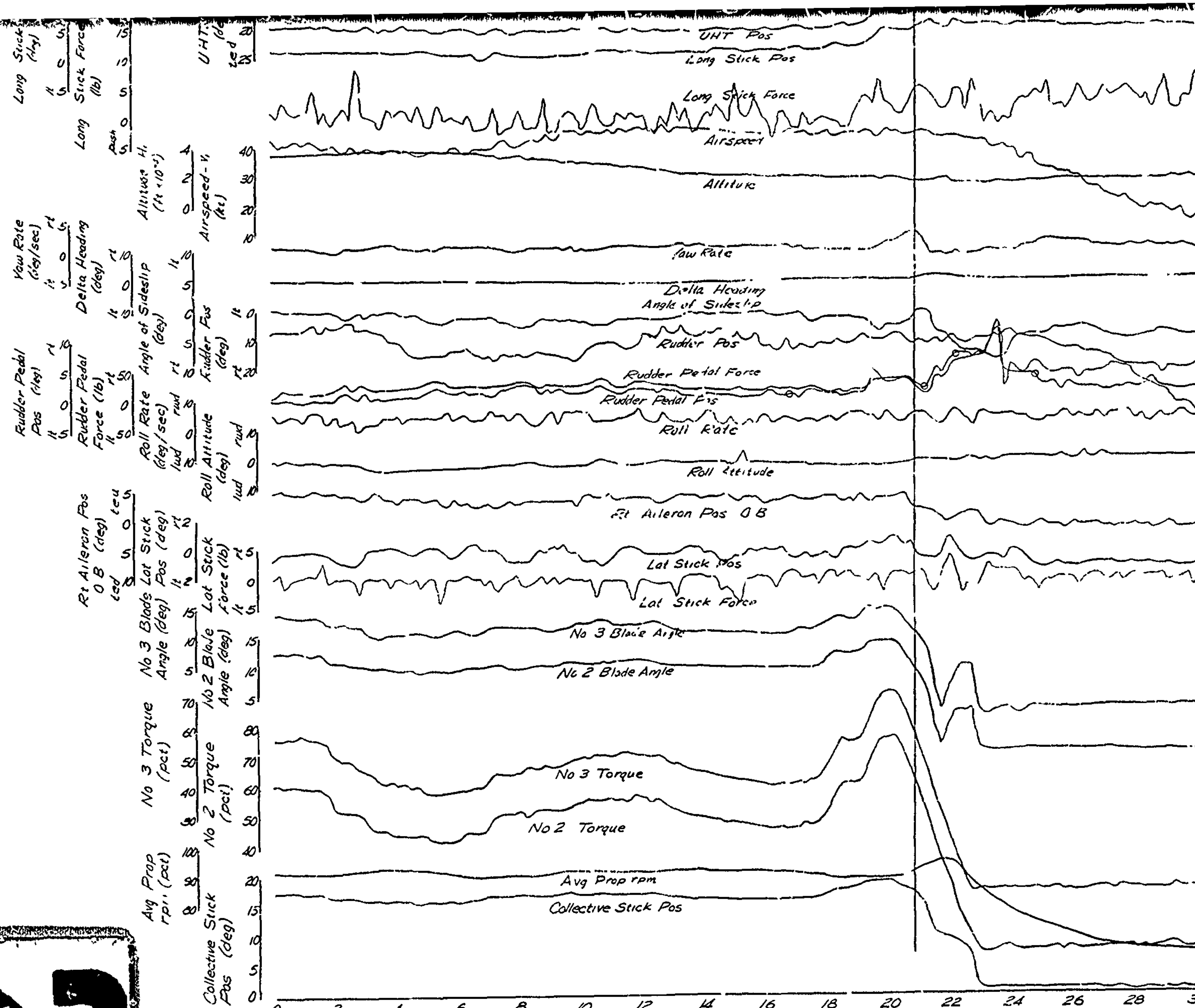
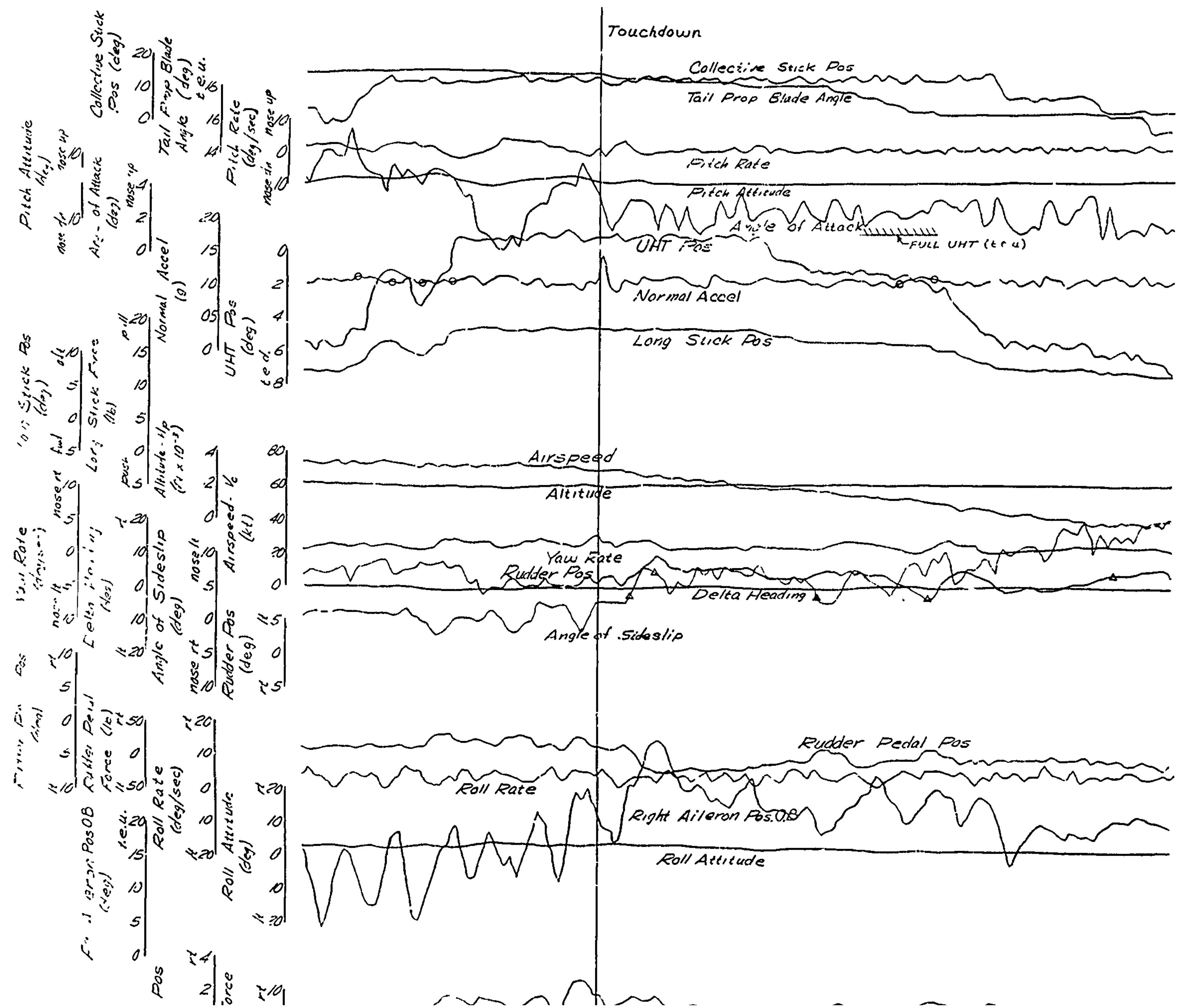


Figure 22. 35°/80° Landing

XC-142A USAF S/N 62-5923
Gross Weight : 36 980 lb cg : 19.08 % MGC
Tail Prop : Disengaged SAS : On



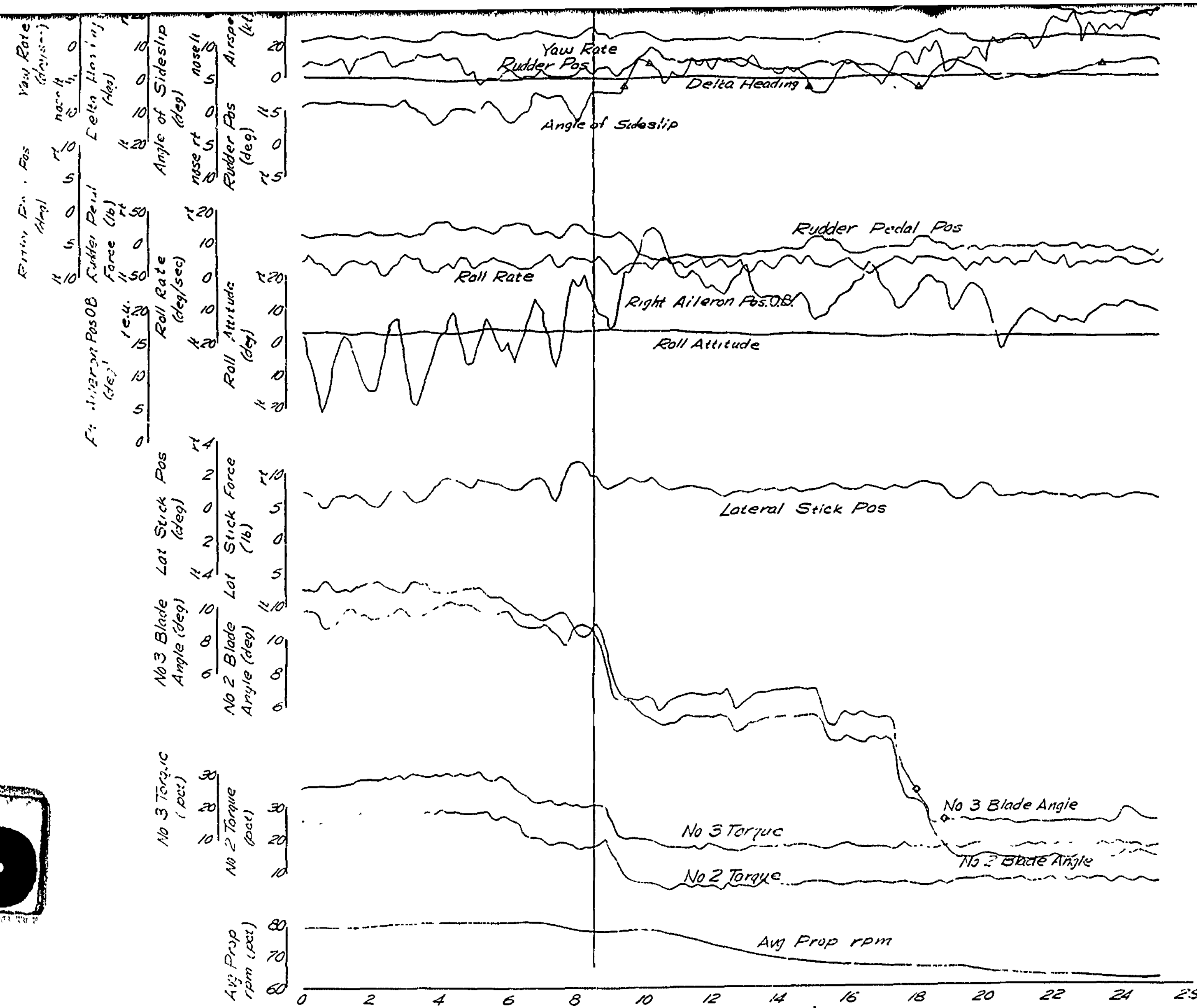
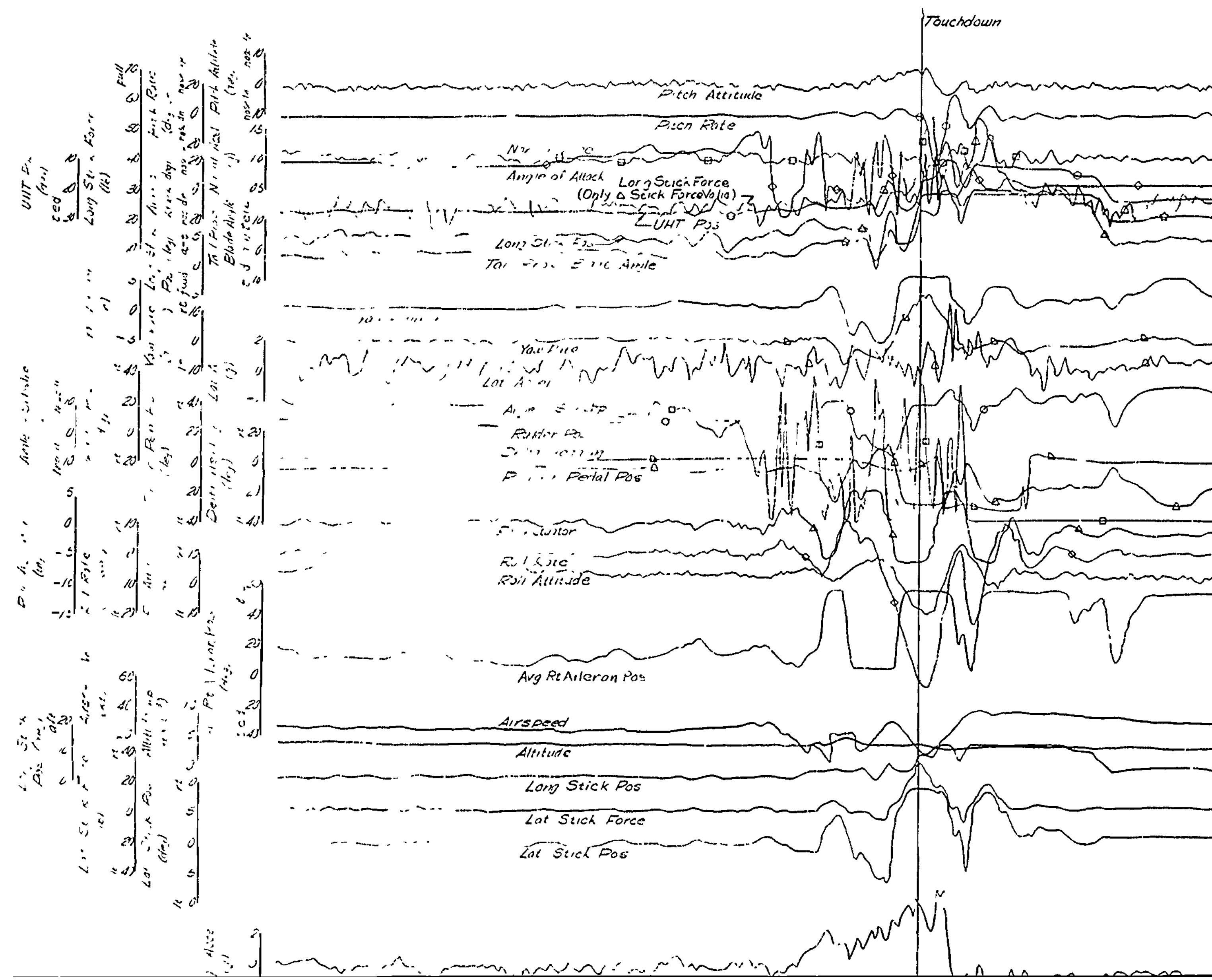


Figure 23. 20°/30° Landing

XC-142 A USAF S/N 62-5922
Gross Weight 37350 cg 18.2 % MGC
Tail Prop: Engaged SAS On



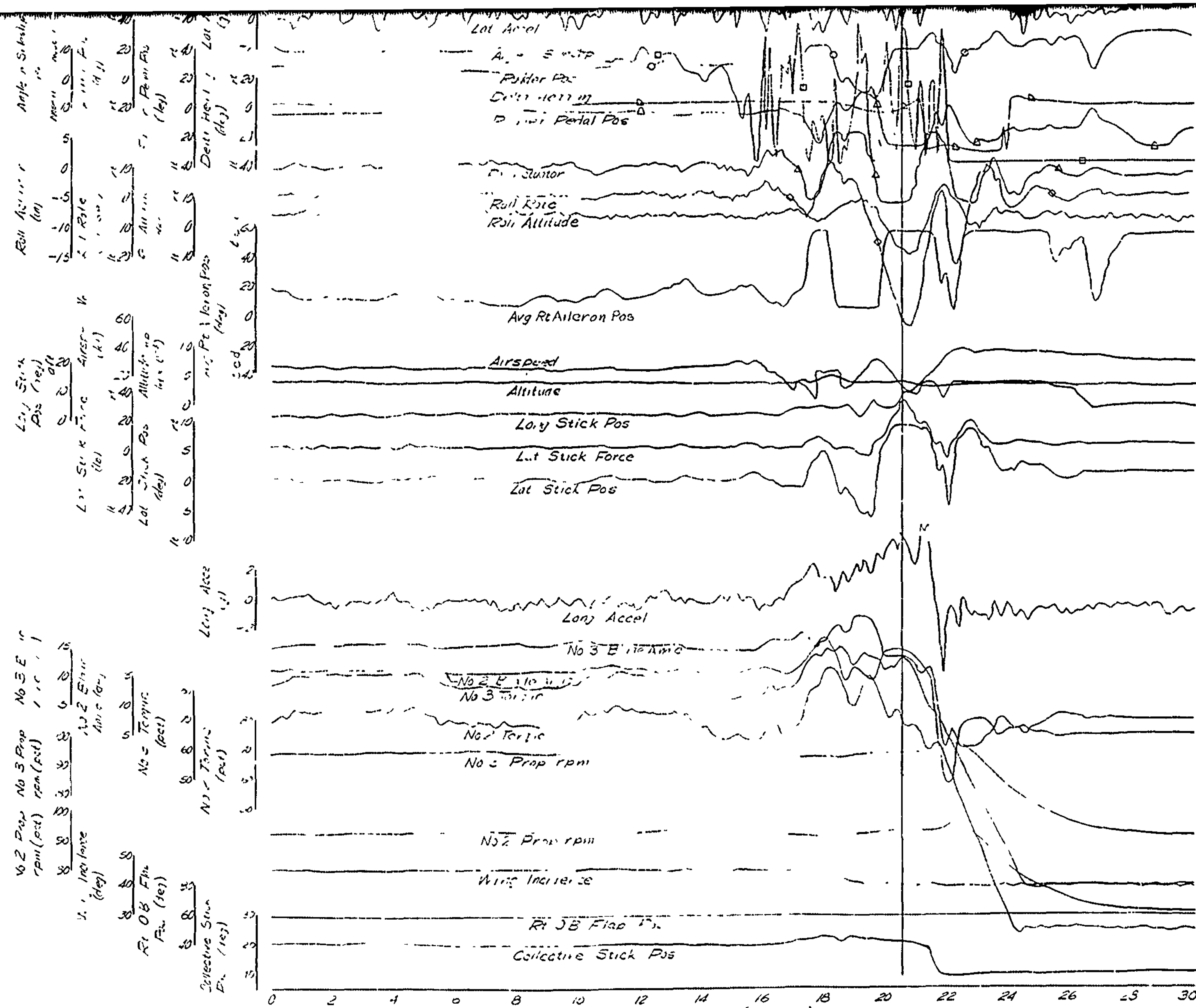
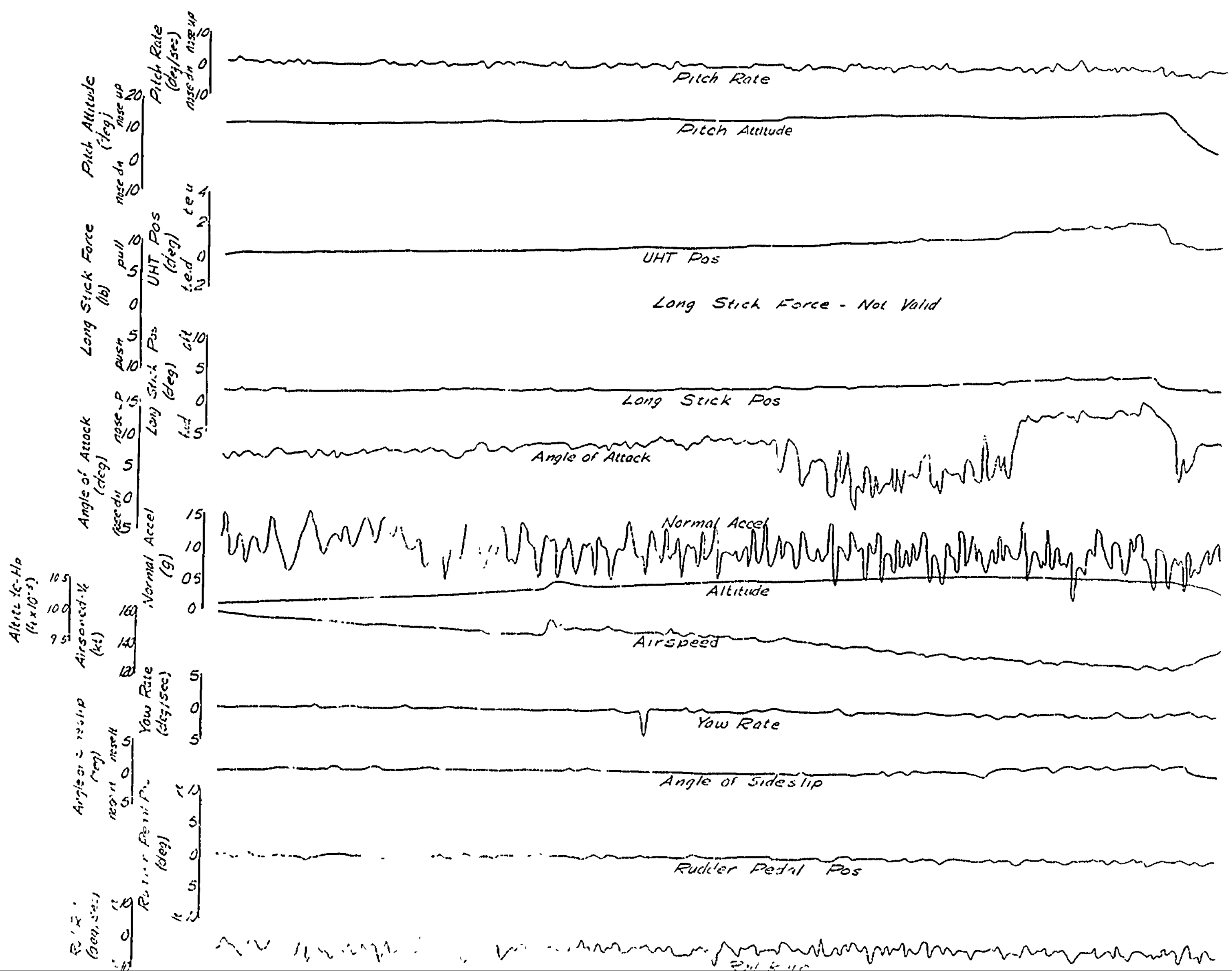


Figure 24 45°/60° Landing

XC-142 A USAF S/N 62-5921
Gross Weight . 36 570 lb cg 20.95 % MGC
Tail Prop : Disengaged SAS Off



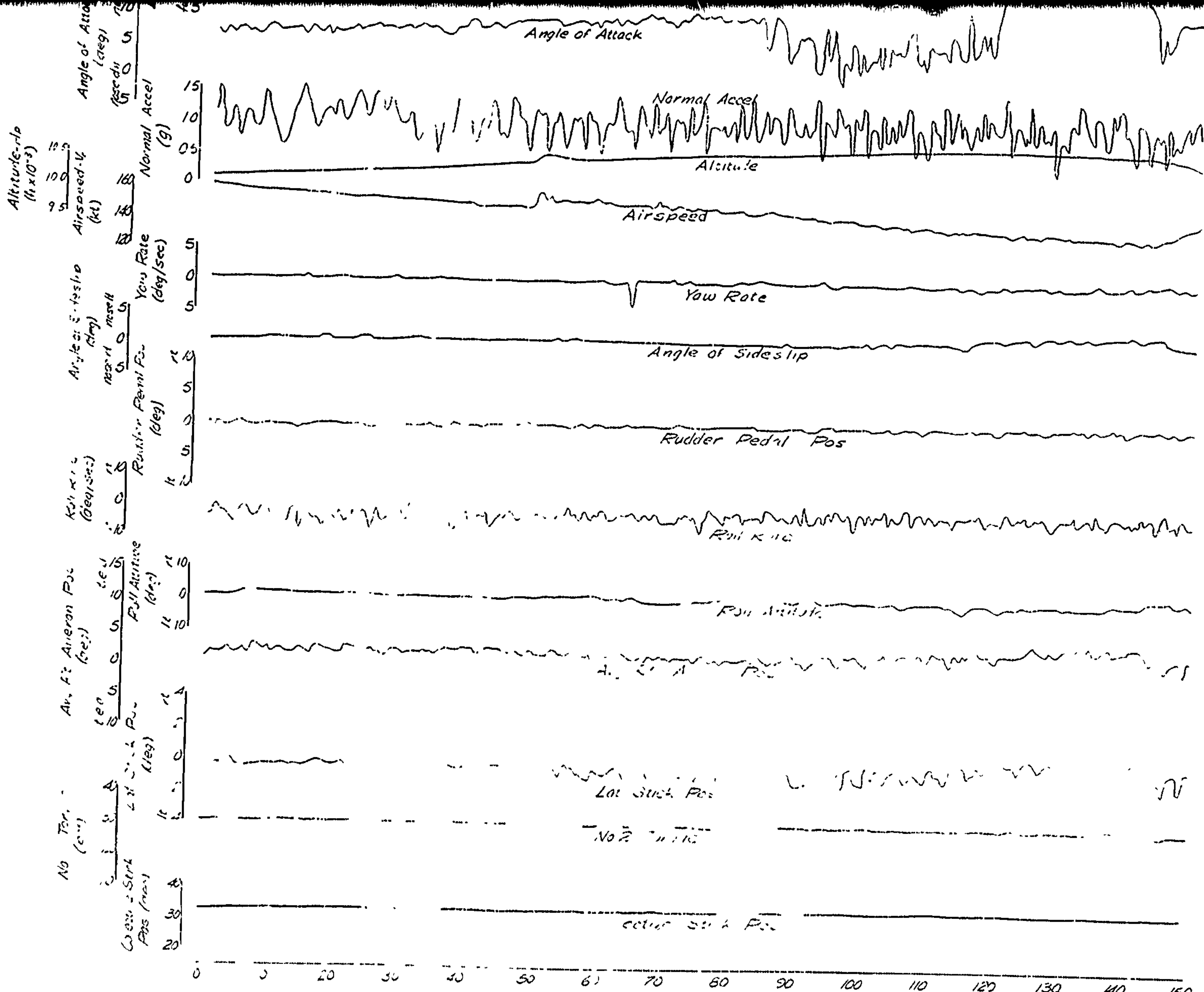
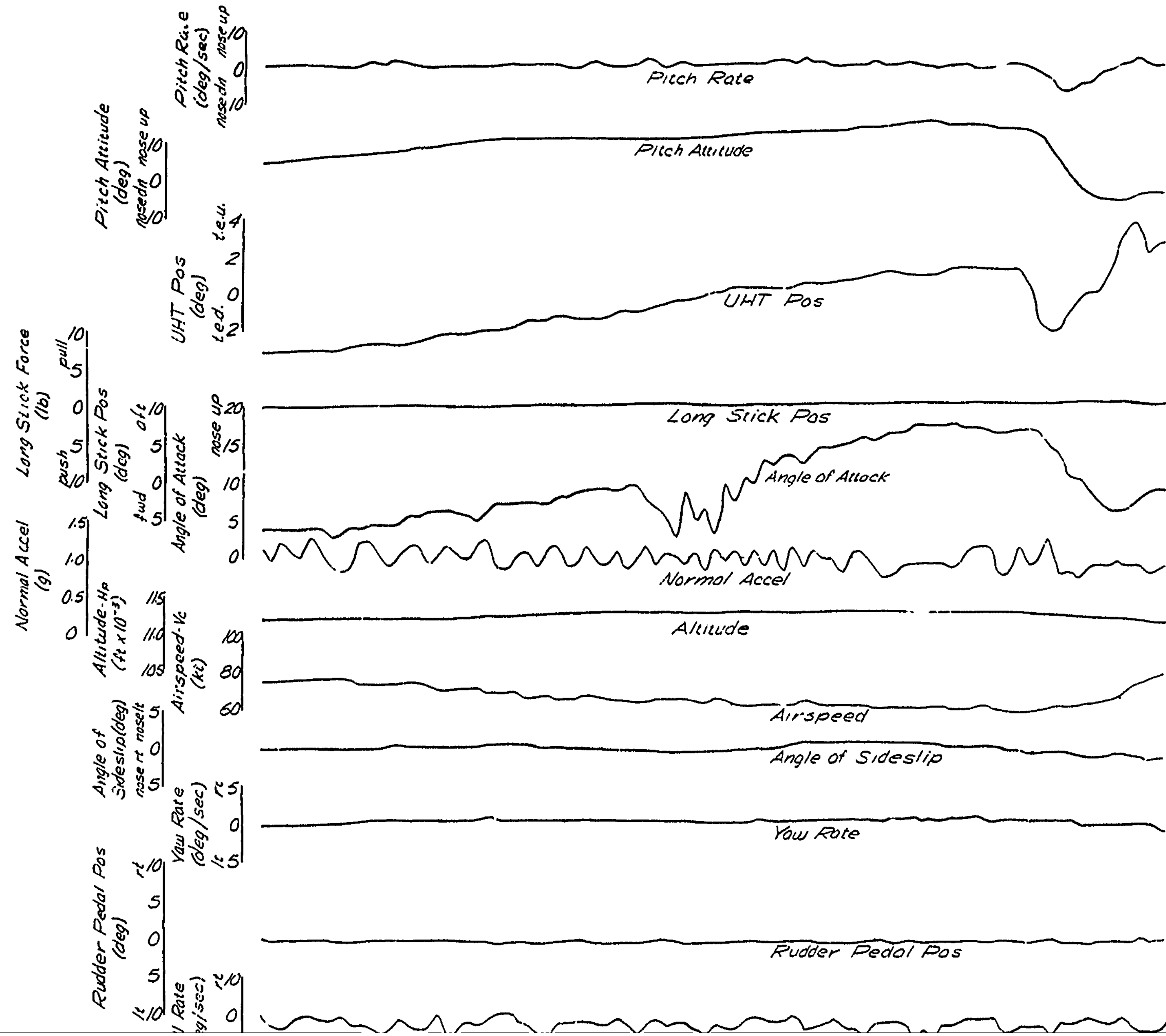


Figure 25 Power On Stall - 0°/0°

XC-142A USAF S/N 62-5921
Gross Weight: 35820 lb cg: 19.73% MGC
Tail Prop: Disengaged SAS: On



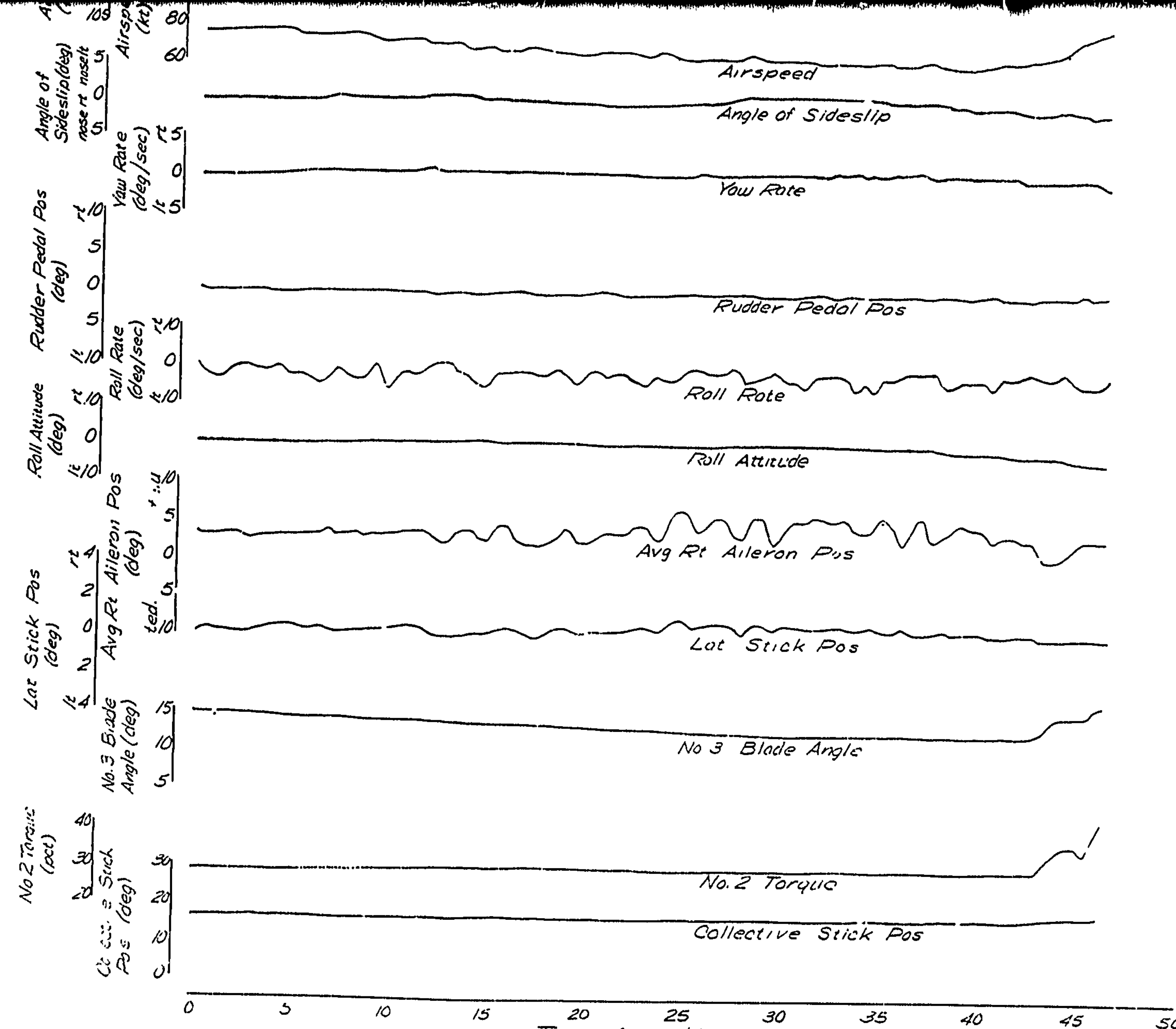
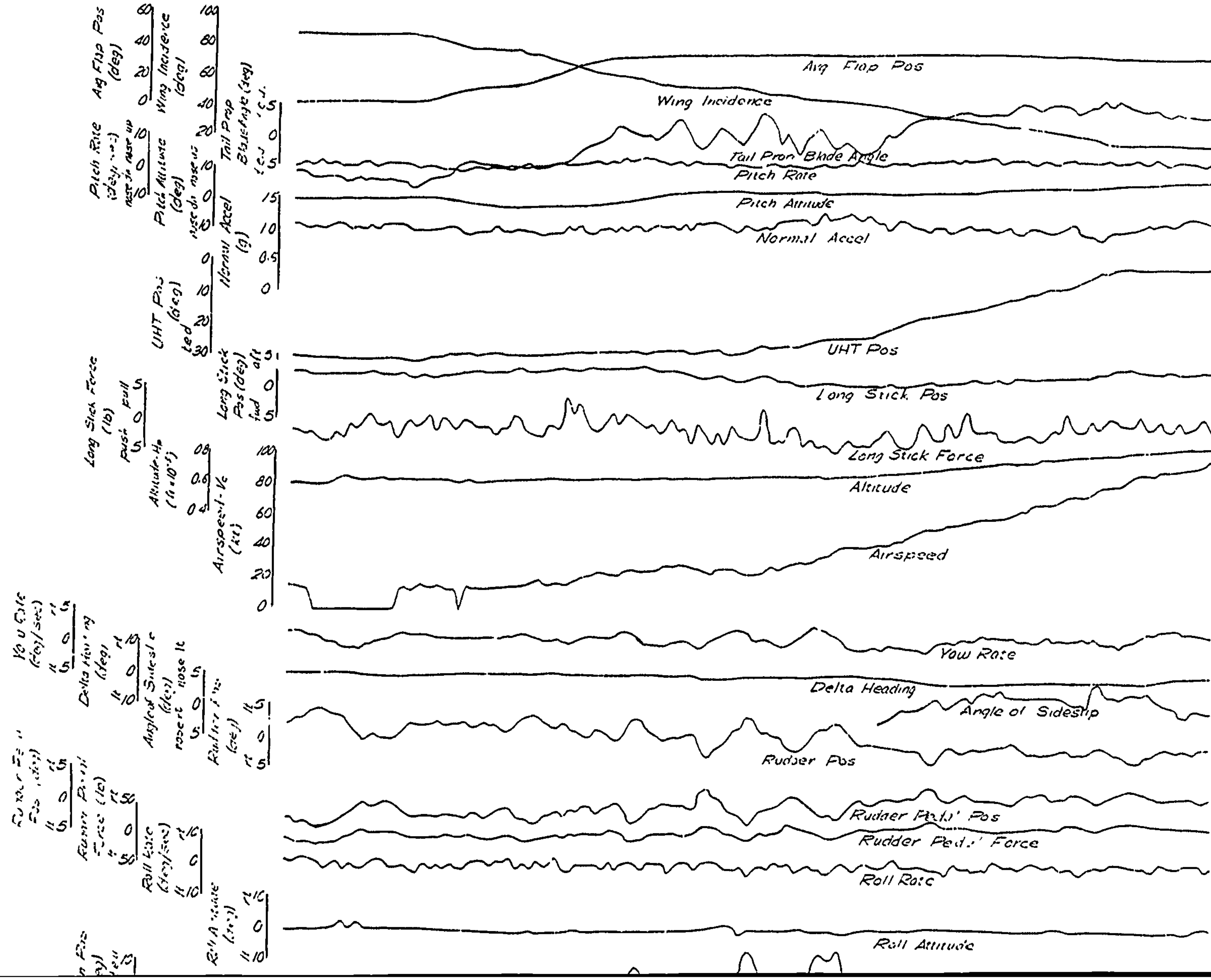


Figure 26 Power On Stalls - 14°/30°

XC-142A USAF S/N 62-5923
Gross Weight · 34 530 lb cg 18.80 % MGC
Tail Prop Engaged SAS · Pitch and Roll: On
Yaw: Off



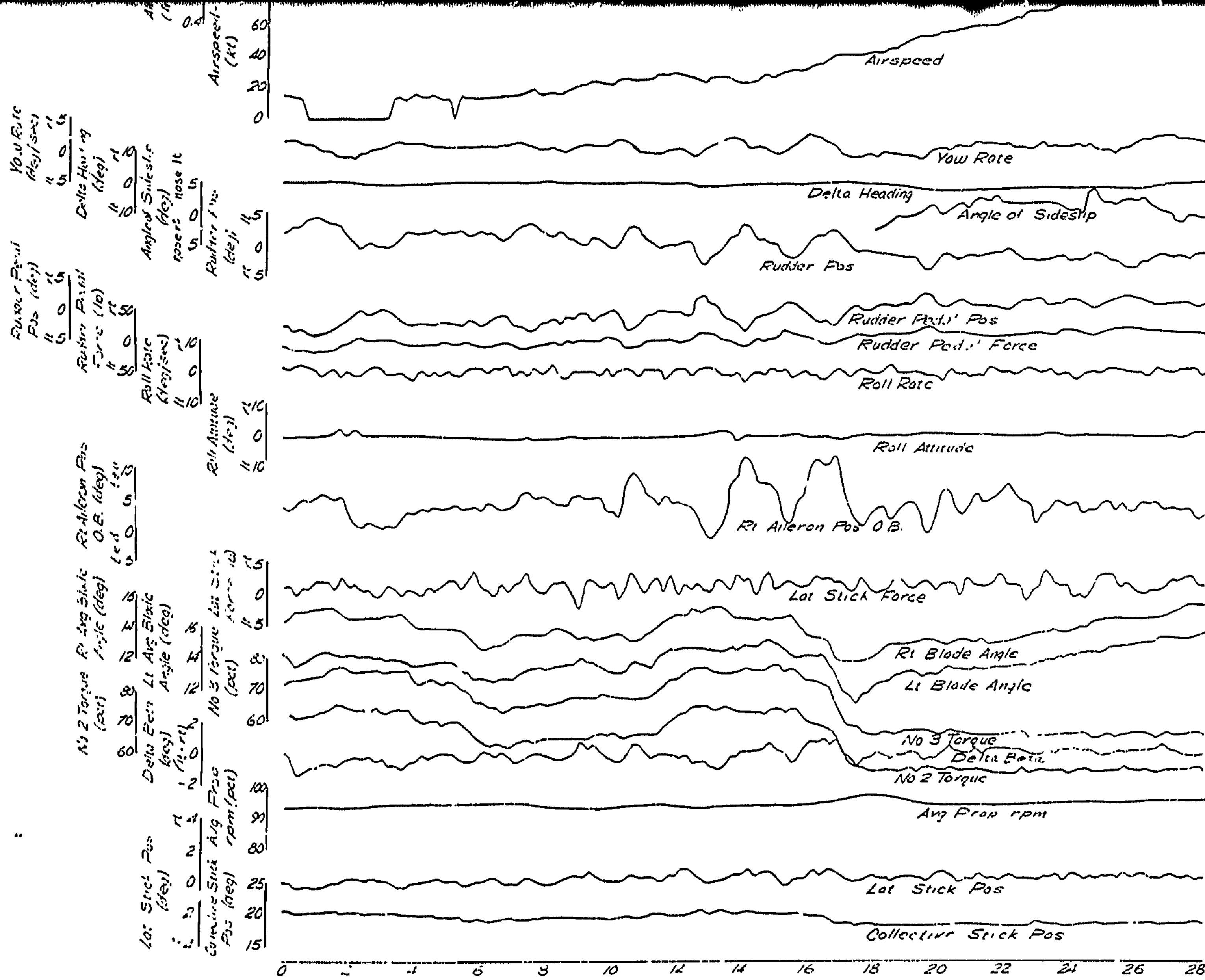
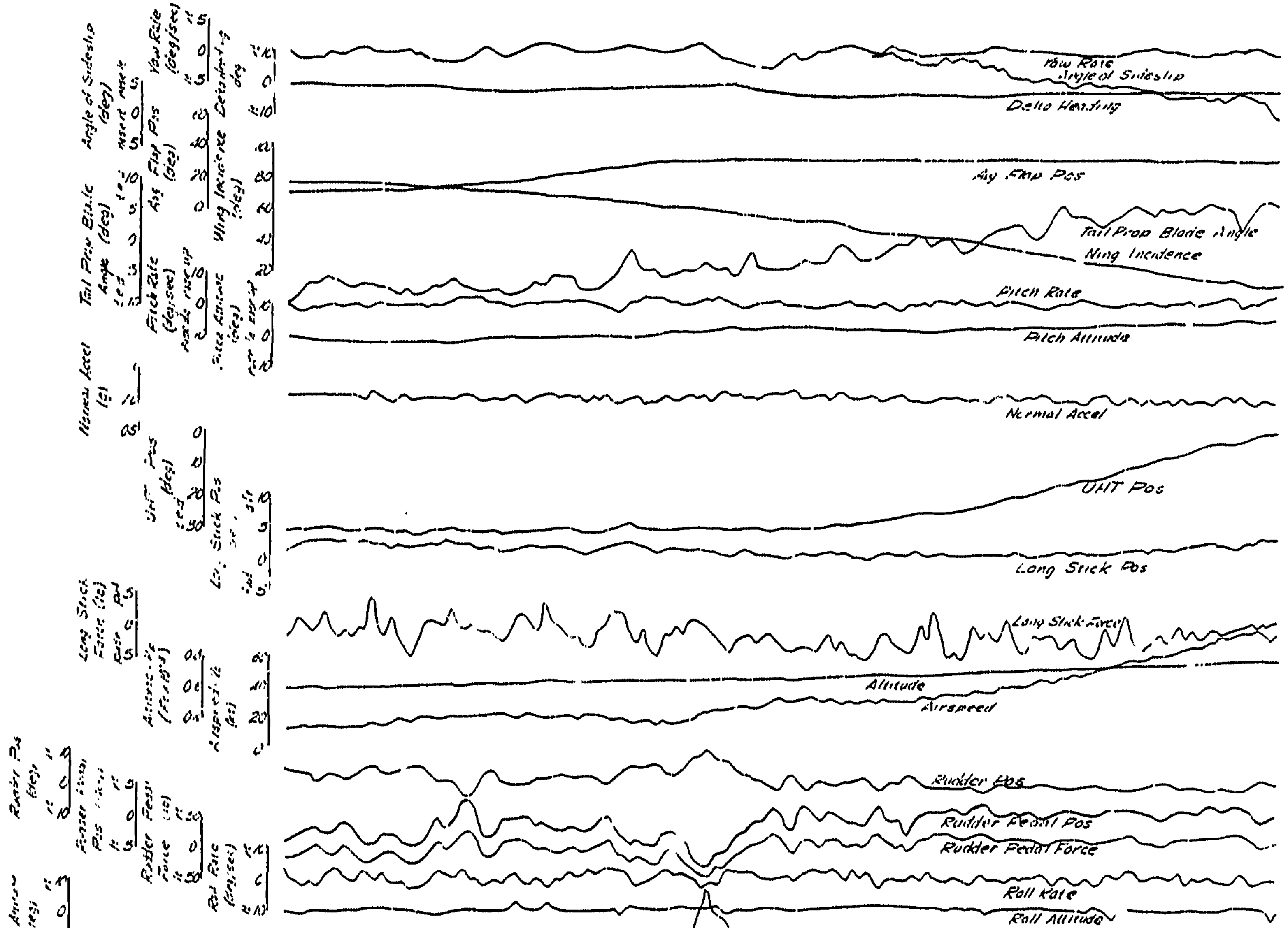


Figure 27 Conversion

XC-142A USAF S/N 62-5925
Gross Weight: 33880 lb cg. 18.46% MAC
Tail Prop. Engaged SAS: Off



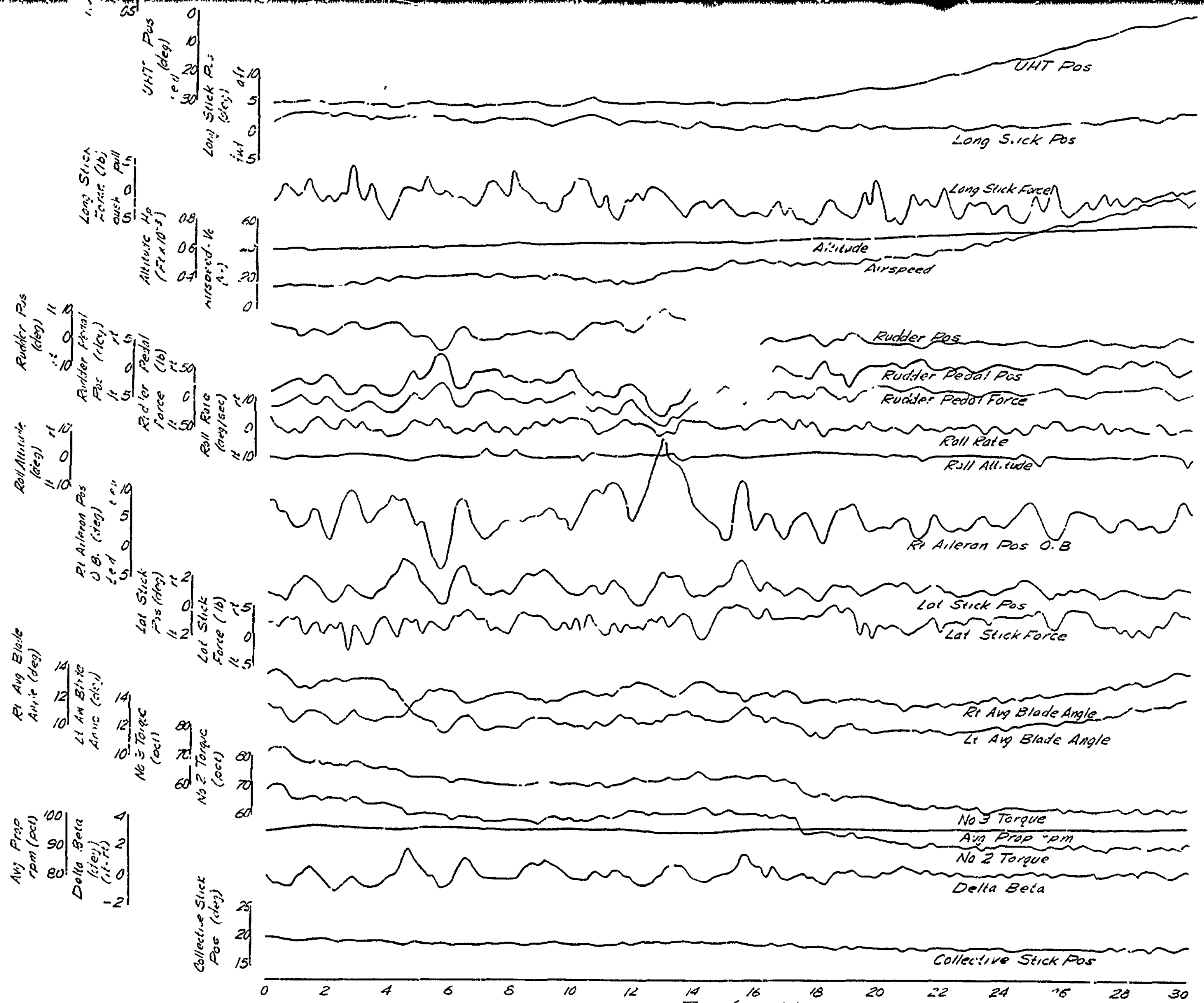
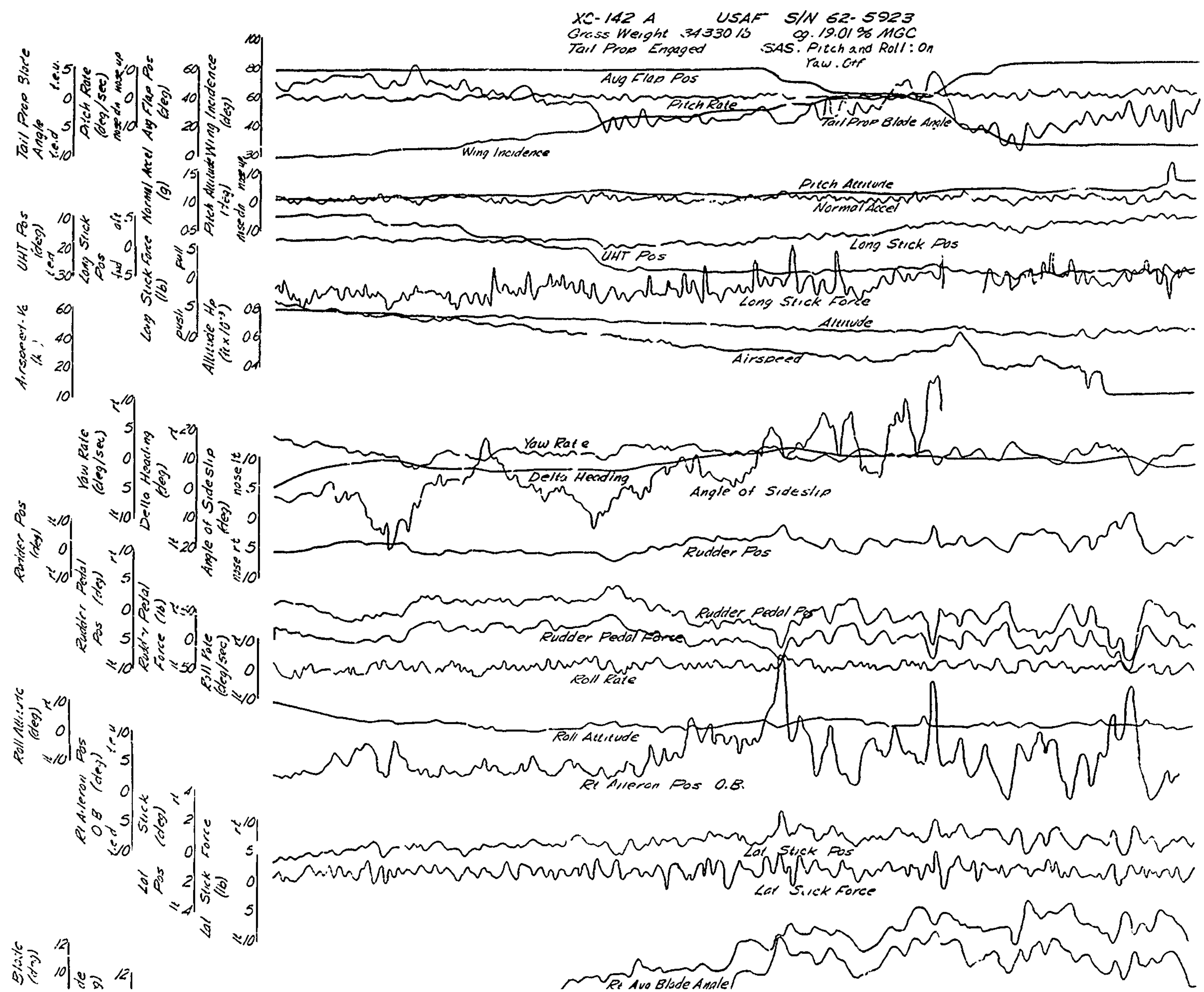


Figure 28 Conversion



XC-142 A USAF S/N 62-5923
Gross Weight 34330 lb cg. 19.01 % MGC
Tail Prop Engaged SAS. Pitch and Roll: On
Yaw: Off
Aug Flap Pos
Pitch Rate
Tail Prop Blade Angle
Wing Incidence
Pitch Attitude
Normal Accel
Long Stick Pos
UHT Pos
Long Stick Force
Altitude
Airspeed
Yaw Rate
Delta Heading
Angle of Sideslip
Rudder Pos
Rudder Pedal Pos
Rudder Pedal Force
Roll Rate
Roll Attitude
Rt Aileron Pos O.B.
Lat Stick Pos
Lat Stick Force
Rt Avg Blade Angle

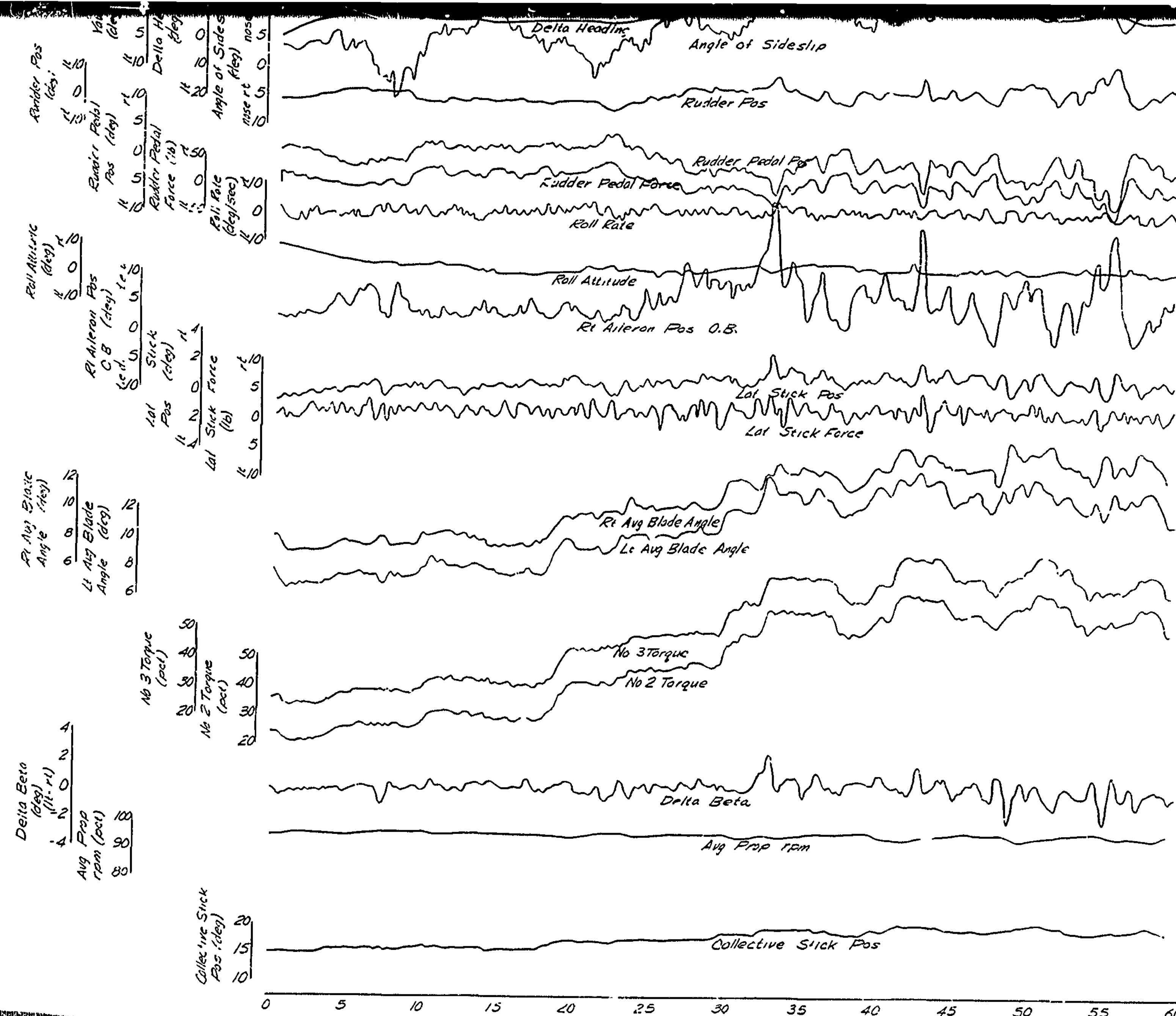
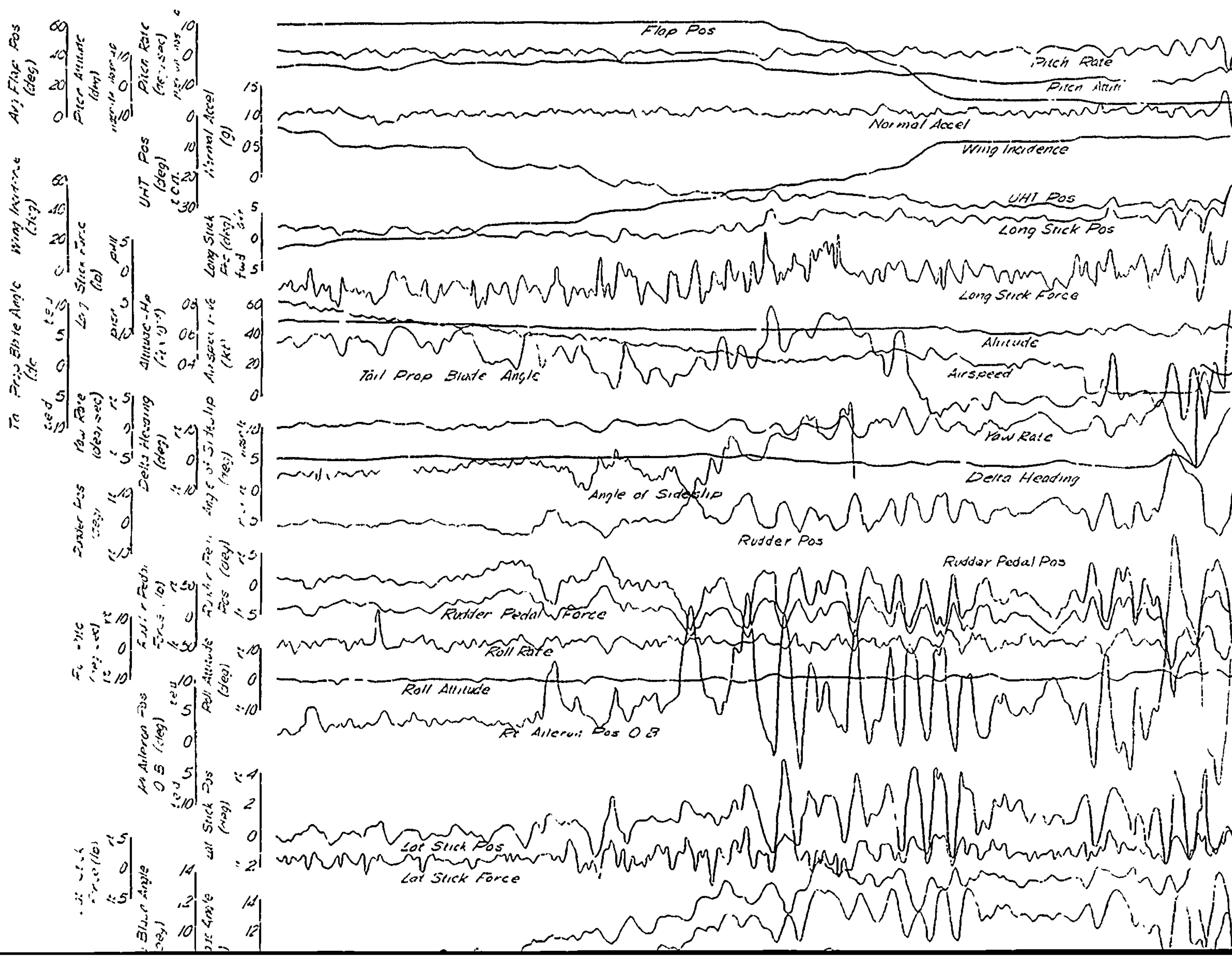


Figure 29 Reconversion

XC-142A USAF S/N 62-5923
Gross Weight . 33730 lb cg. 19.41 % MGC
Tail Prop · Engaged SAS : Off



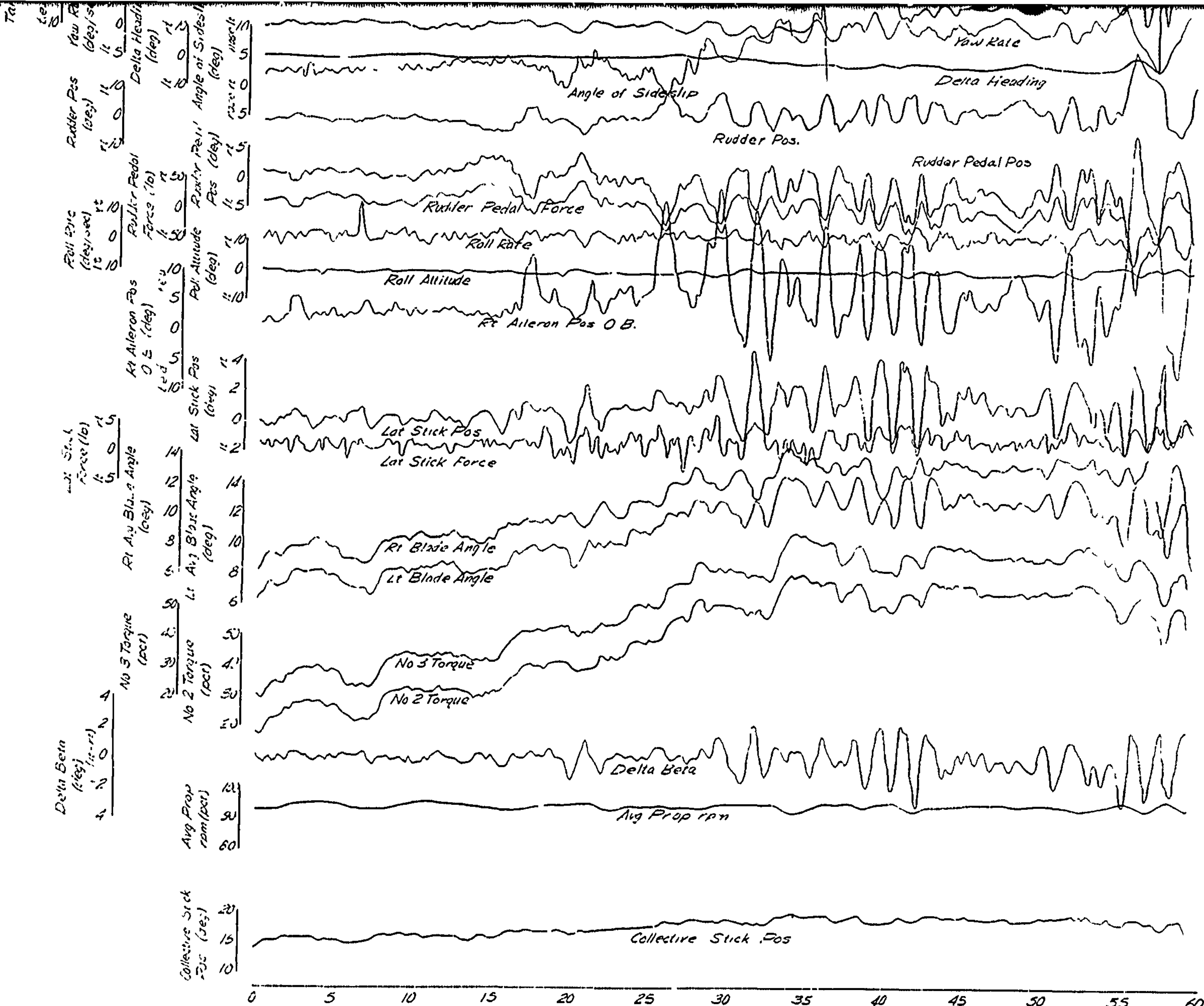
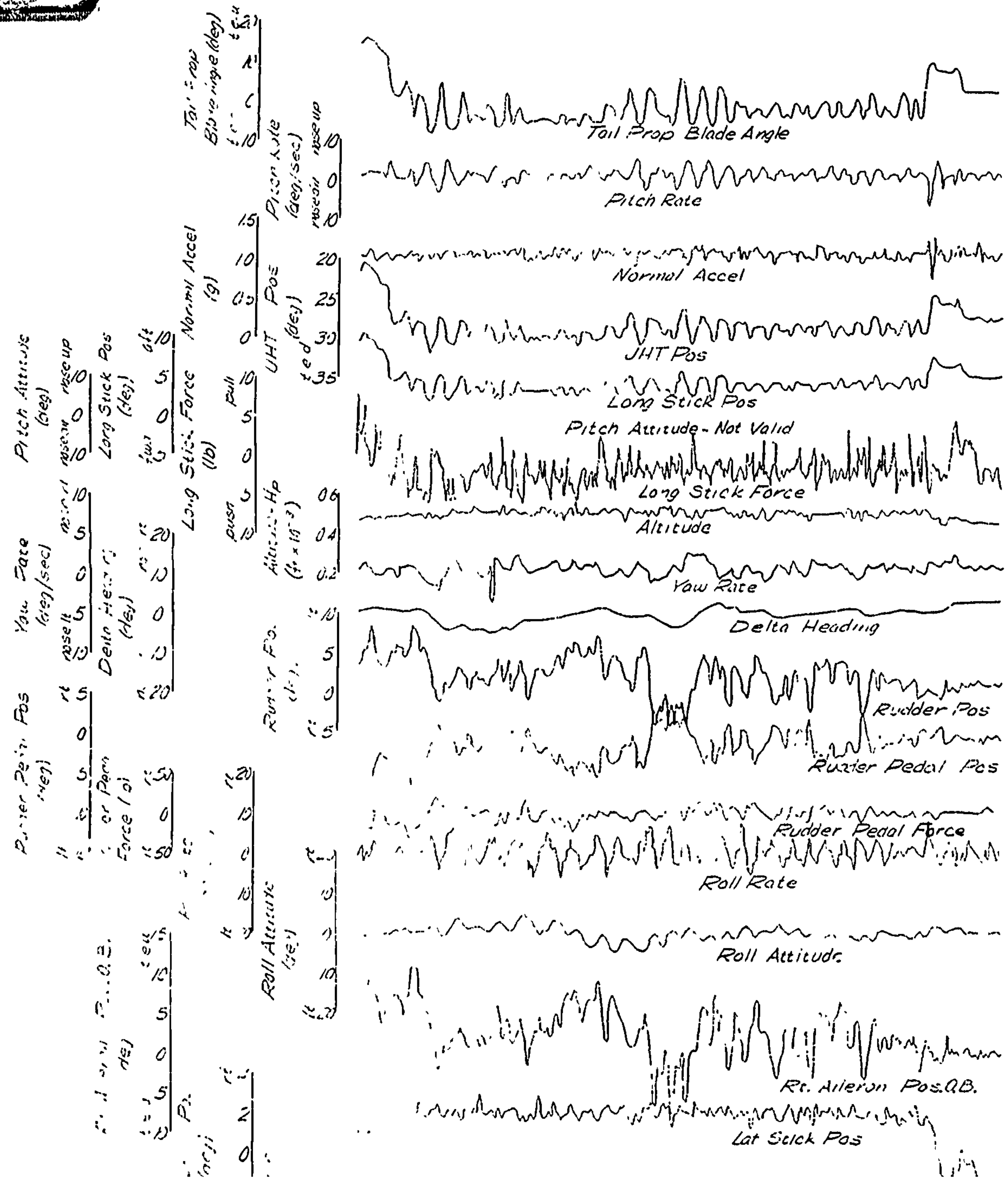


Figure 30. Reconversion

XC-142A USAF S/N 62-5923
Gross Weight: 33430 lb cg 19.10% MGC
Tail Prop: Engaged SAS. Off



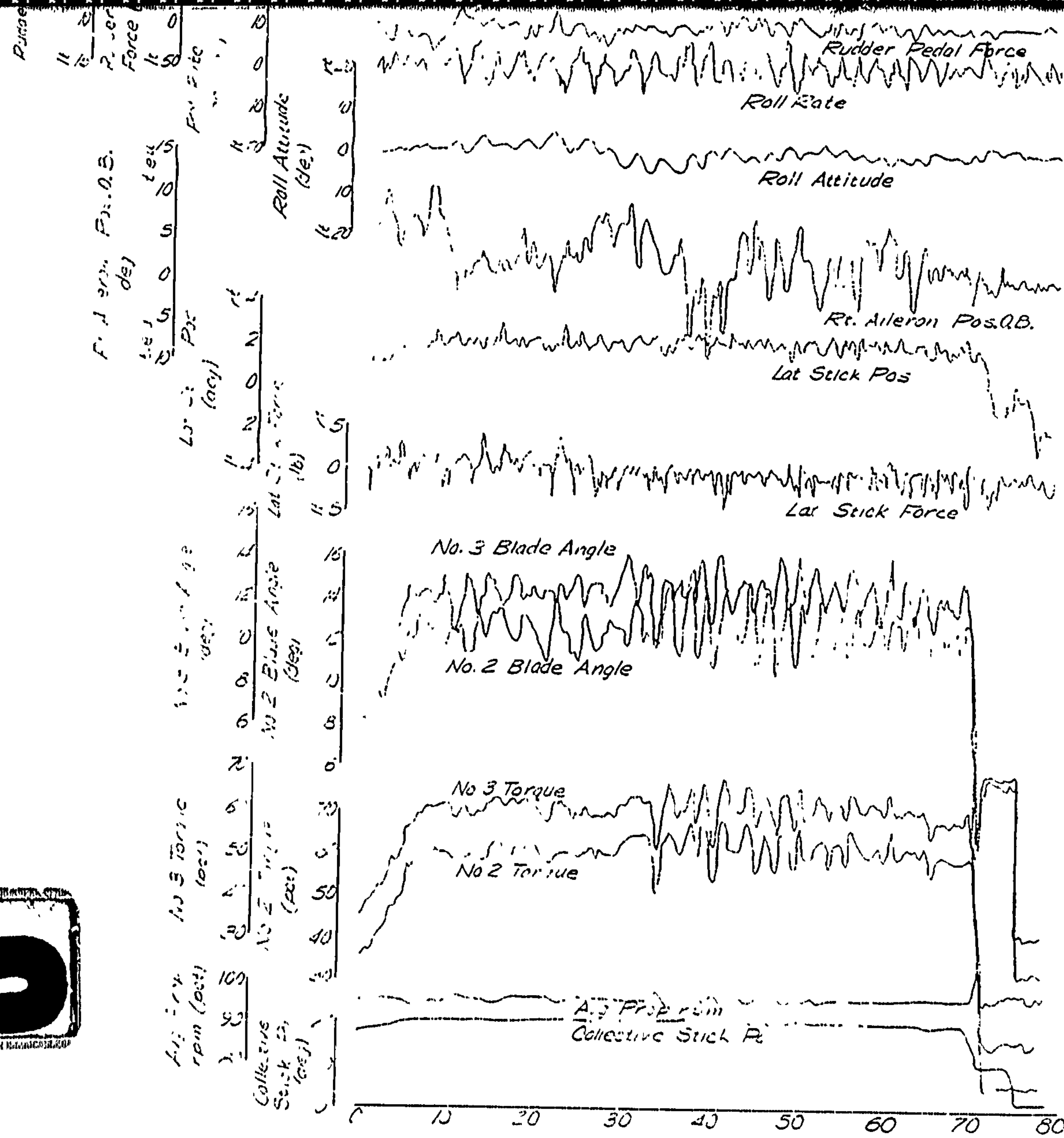

Rudder Pedal Force
Roll Rate
Roll Attitude
Rt. Aileron Pos.Q.B.
Lat Stick Pos
Lat Stick Force
No. 3 Blade Angle
No. 2 Blade Angle
No 3 Torque
No 2 Torque
Collective Stick P.
0
10
20
30
40
50
60
70
80

Figure 31

APPENDIX II

GENERAL AIRCRAFT INFORMATION

WEIGHT AND BALANCE

XC-142A USAF S/N 62-5923:

Empty weight (including unusable fuel and oil)	24 125 lb
Crew (pilot and copilot)	400 lb
Test instrumentation	2 427 lb
Empty water ballast tanks	2 670 lb
Trapped water ballast	475 lb
"Permanent" ballast	229 lb
Dry weight	30 326 lb
Center of gravity	23.45 pct MGC

AIRCRAFT FLIGHTS LIMITS[5]

Gross Weight:

Maximum takeoff	41 500 lb
Maximum landing	37 474 lb

Center of Gravity (Wing Down):

Forward	21 percent MGC
Aft	21 percent MGC

Note:
The limits were increased by the contractor near the end of this evaluation to 17 percent MGC forward and 28 percent MCC aft.

Load Factor:

Symmetrical	0 to +2.0 g
Unsymmetrical	±1 g

Airspeed and Altitude:

V_{max} = 245 KEAS (80 percent limit load factor)

Landing gear extension	170 KIAS
Landing gear retraction	140 KIAS

[5] These limits were set by the contractor for the MPP tests, based on their flight envelope at that time. These were not the aircraft design limits.

Flaps	10 deg	185 KEAS	
	20 deg	170 KEAS	
	30 deg	150 KEAS	
	40 deg	140 KEAS	
	60 deg	72 KEAS	
Wing above zero degrees			140 KEAS
Wing unlocked			170 KEAS
Tail propeller clutch/declutch			110 to 150 KEAS
Cargo door (do not operate in-flight)			
Maximum altitude			25 000 feet
Maximum rearward velocity (VTOL)			10 KTAS
Maximum sideward velocity (VTOL)			10 KTAS

Landing:

Maximum wing angle for takeoff or landing	35 deg
Maximum crosswind component	10 KTAS
Maximum touchdown rate of sink	10 fps at 37 474 lb

Taxi:

Minimum wing angle	10 deg

AIRCRAFT DIMENSIONAL DATA

This data was taken from reference 3.

Wing:

Total wing area	534.37 sq ft
Span	67.55 ft
Aspect ratio	8.53
Taper ratio	0.61
Sweepback of 1/4 chord	4.13 deg
Dihedral angle (root chord at B.L. = 0 to tip chord)	-2.12 deg
Geometric twist	0.0
Root chord	9.83 ft
Tip chord	6.0 ft
Mean geometric chord	
Length	8.072 ft
Spanwise location from B.L. = 0	15.51 ft
Location of leading edge from nose of fuselage	235.47 in.
Airfoil section	NACA 63-318

Leading Edge Slats:

Spanwise location (percent semispan)	
Inboard	
Root	44.98
Tip	68.90
Outboard	
Root	87.39
Tip	97.84

Area (square feet)	
Inboard (each)	10.34
Outboard (each)	2.56

Trailing Edge Flaps:

Type	Double Slotted
Flap span (percent wing semispan)	
Inboard	
Root	15.43
Tip	35.74
Center	
Root	43.60
Tip	78.25
Outboard	
Root	86.08
Tip	98.92
Flap chord, in flap chord plane (percent wing chord)	33.0
Flap area, in flap chord plane (square feet)	
Inboard (one side)	19.09
Center, including aileron (one side)	29.00
Outboard, including aileron (one side)	9.10
Total (one side)	57.19
Maximum deflection	60 deg

Ailerons:

Type	Plain
Spanwise location (percent semispan)	
Center	
Root	43.30
Tip	78.40
Outboard	
Root	86.03
Tip	99.61
Area aft of aileron hinge line (square feet)	
Center (per side)	22.28
Outboard (per side)	7.11
Total (per side)	29.39
Maximum deflection (per side)	
Cruise	+17.5 deg
Hover	$\mp$50.0 deg
Transition: deflection programed with wing incidence.	
Down aileron deflection + flap deflection	60 deg
Up aileron deflection	50 deg

Horizontal Stabilizer:

Type	All Moving Unit Hor. Tail
Total horizontal stabilizer	163.5 sq ft
Span	31.14 ft
Aspect ratio	5.93
Taper ratio	0.5

Sweepback of 1/4 chord	12.66 deg
Dihedral angle (root chord at B.L. = 0 to tip chord)	0.0
Geometric twist	0.0 deg
Root chord	7.0 ft
Tip chord	3.5 ft
Mean geometric chord	
Length	5.44 ft
Spanwise location from B.L. = 0	6.92 ft
Fuselage station of leading edge	549.67
Maximum deflection, cruise	
Leading edge up	5.0
Leading edge down	9.0
Maximum deflection, transition, static (deg)	
Leading edge up	39.0
Leading edge down	9.0
Airfoil section	
Root	NACA 0015
Tip	NACA 0012

Vertical Tail:

Area (including rudder)	130.0 sq ft
Span	15.625 ft
Aspect ratio	1.88
Taper ratio	.25
Sweepback of 1/4 chord	26.0 deg
Root chord	13.32
Tip chord	3.33
Mean geometric chord	
Length	9.33 ft
Spanwise location from W.L. 100	10.21 ft
Fuselage station of leading edge	498.0
Air foil section	
Root	NACA 0018
Tip	NACA 0012

Rudder:

Type	Plain
Spanwise location (percent vertical tail span)	
Root	41.6
Tip	100.0
Chord (percent vertical tail chord)	
Root	40
Tip	40
Mean geometric chord	2.68 ft
Area, aft of hinge line	22.82 sq ft
Maximum deflection	±30 deg

Fuselage:

Length	50.0 ft
Maximum height	10.38 ft
Maximum width	9.25 ft

Maximum cross-sectional area	92.0 sq ft

Main Propellers:

Designation	2FE16A3-4A and 2FE16A4-4A
Number of blades	4
Diameter	15.625 ft
Activity factor per blade	86.0
Integrated design lift coefficient	0.475
Area (per propeller)	191.75 sq ft

Tail Propeller:

Designation	1FA9A4-10A
Number of blades	3
Diameter	8.167 ft
Activity factor per blade	145
Integrated design lift coefficient	0

Wetted Area:

Fuselage	1652 sq ft
Nacelles, four	446 sq ft
Wing, total	772 sq ft
Vertical tail	222 sq ft
Horizontal tail	320 sq ft
Miscellaneous	85 sq ft
Total	3447

APPENDIX III

TEST DATA CORRECTED FOR INSTRUMENT ERROR

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 23 Run 1
Date 27 MAR 65
Pressure Altitude 203 Ft
Temperature 5 0 °C
Gross Weight 41 300 Lb

	LO/TD	50 Ft
V_{gr}	38.4 Kt	55 1 Kt

Wind 9 9 Kt at 50°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
89	22 9							
162	45 0							
223	67 9							
275	94 1							
309	124 3							
348	147 5							
385	177 1							
419	200 0							
457	223 6	8						
489	246 7	1 6						
521	268 9	6 6						
551	290 0	9 9						
579	312 6	14 9						
614	334 2	19 8						
645	357 3	26 4						
672	379 5	33 0						
700	402 3	40 3						
728	426 9	48 6						
758	452 0	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 24 Run 1
Date 27 MAR 65
Pressure Altitude 225 Ft
Temperature 5 5 °C
Gross Weight 40 300 Lb

	LO/TD	50 Ft
V_{gr}	37.4 Kt	51 2 Kt

Wind 7 04 Kt at 60°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
80	13 9							
165	36 9							
222	59 9							
274	85 3							
317	109 9							
356	135 0							
395	155 3	8						
430	176 1	3 3						
463	196 9	4 9						
492	228 6	8 2						
529	258 3	2 3						
560	279 3	17 2						
588	303 3	19 7						
619	328 9	25 4						
651	356 6	31 1						
681	391 7	36 8						
710	406 5	41 0						
739	432 1	47 5						
768	442 .	53 3						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 24 Run 3
Date 27 MAR 65
Pressure Altitude 230 Ft
Temperature 6 0 °C
Gross Weight 40 100 Lb

	LO/TD	50 Ft
V_{gr}	44 6 Kt	53 9 Kt

Wind 6.6 Kt at 60°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
73	223							
147	489							
208	773							
259	105 5							
293	126 0							
334	153 6							
369	185 0							
403	205 6	3 3						
441	233 6	5 8						
473	258 3	9.1						
505	282 2	10 7						
534	306 1	14.8						
563	328 6	18.1						
598	358 1	24 7						
629	381 7	30 5						
666	403 9	36.3						
694	424 6	42 9						
722	444 3	48 6						
752	465 1	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 24 Run 5
Date 27 MAR 65
Pressure Altitude 235 Ft
Temperature 6 3 °C
Gross Weight 39 700 Lb

	LO/TD	50 Ft
V_{gr}	38 2 Kt	53 4 Kt

Wind 4 2 Kt at 120°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
96	22 8							
195	51 4							
276	79 2							
336	--							
355	—							
398	—							
421	—							
464	198 0	0						
506	220 9	3 2						
536	246 2	4 6						
570	273 2	6 6						
600	298 5	9 8						
629	325 5	12 3						
659	351 7	17 2						
687	376 2	22 1						
714	402 4	26 2						
739	430 2	32 0						
767	455 5	38 5						
796	480 8	43 3						
821	492 0	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 24 Run 7
Date 27 MAR 65
Pressure Altitude 240 Ft
Temperature 7 6 °C
Gross Weight 37 150 Lb

	LO/TD	50 Ft
V_{gr}	43 2 Kt	58 6 Kt

Wind 5 63 Kt at 100°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
88	20 5							
1 71	53 3							
2 16	73 8							
2 57	95 1							
2 87	114 8							
2 31	141 0							
2 66	164 8							
305	192 7							
334	2130	2 46						
369	2409	4 92						
399	2647	656						
331	2893	901						
461	3139	12 30						
489	3385	16 40						
5 16	3615	20 5						
545	3869	25 4						
570	4099	29 5						
600	4361	36 3						
6 24	4591	42 6						
651	4921	47 8						
662	5070	500						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 24 Run 9
Date 27 MAR 65
Pressure Altitude 240 Ft
Temperature 8 0 °C
Gross Weight 38700 Lb

	LO/TD	50 Ft
V_{gr}	44 2 Kt	53 1 Kt

Wind 6 6 Kt at 100°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
63	22 8							
151	49 2							
212	75 6							
26?	100 4							
299	125 2							
323	141 7							
361	168 1							
395	192 4							
427	2200	5						
458	2425	34						
488	2611	66						
516	2920	9 9						
545	3176	13 2						
572	340 6	16 5						
601	366 2	231						
627	389 2	260						
655	414 0	275						
684	4396	305						
7 11	462 6	354						
735	481 4	376						
7 66	513 6	46 2						
793	530 0	500						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 25 Run 1
Date 31 MAR 65
Pressure Altitude 120 Ft
Temperature 13 3 °C
Gross Weight 41 150 Lb

	LO/TD	50 Ft
V_{gr}	40 5 Kt	51 6 Kt

Wind 7 9 Kt at 90°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
-	-							
1 50	58							
2 02	85							
2 45	109							
2 92	139							
3 33	167	5						
3 68	189	1 7						
4 00	213	2 5						
4 33	238	4 1						
4 64	252	4 9						
4 97	287	8 1						
5 30	313	1'5						
5 61	340	15 1						
5 91	365	19 8						
6 32	389	24 9						
6 59	413	31 3						
6 80	439	37 1						
7 11	46 5	43 6						
7 38	490	46 5						
7 67	502	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 25 Run 3
Date 31 MAR 65
Pressure Altitude 125 Ft
Temperature 13 2 °C
Gross Weight 40900 Lb

	LO/TD	50 Ft
V_{gr}	41 1 Kt	50 0 Kt

Wind 7 8 Kt at 90°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0		0						
-		0						
1 60	70	0						
2 03	90	0						
2 52	119	0						
2 93	145	0						
3 31	171	5						
3 66	195	9						
4 00	220	16						
4 33	246	2 4						
4 64	271	3 3						
4 93	294	6 6						
5 24	319	10 7						
5 56	345	14 3						
5 86	369	20 1						
6 17	395	28 0						
6 47	421	36 4						
6 77	444	43 2						
7 05	479	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 25 Run 5
Date 31 MAR 65
Pressure Altitude 125 Ft
Temperature 13.2 °C
Gross Weight 37 940 Lb

	LO/TD	50 Ft
V_{gr}	37 9 Kt	50 9 Kt

Wind 2 7 Kt at 90° Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0							
–	–							
1 60	64							
2 10	89							
2 50	113	0						
2 90	138	4						
3 28	164	8						
3 62	189	3 2						
3 98	2 15	4 9						
4 29	2 39	7 0						
4 62	2 66	10 7						
4 91	289	14 1						
5 22	313	19 8						
5 33	339	23 9						
5 83	365	30 6						
6 10	387	36 4						
6 39	412	45 5						
6 49	438	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 25 Run 7
Date 31 MAR 65
Pressure Altitude 125 Ft
Temperature 14 6 °C
Gross Weight 35 490 Lb

	LO/TD	50 Ft
V_{gr}	36 6 Kt	47 2 Kt

Wind 2 3 Kt at 90° Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0	0						
0		0						
1 55	65	0						
2 07	91	0						
2 49	116	4						
2 84	140	18						
3 30	164	3 3						
3 69	191	5 0						
4 65	217	6 8						
4 37	241	12 4						
4 58	264	18.2						
5 00	289	24 0						
5 34	315	31 7						
5 67	342	39 6						
6 02	370	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 25 Run 9
Date 31 MAR 65
Pressure Altitude Ft
Temperature 14 6 °C
Gross Weight 35240 Lb

	LO/TD	50 Ft
V_{gr}	38 3 Kt	51 3 Kt

Wind 2 6 Kt at 90° Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0	0						
–		0						
1 46	63	0						
1 91	88	0						
2 31	114	5						
2 72	140	12						
3 06	161	2 5						
3 38	186	5 1						
3 67	209	8 3						
3 98	233	11 6						
4 29	259	16 5						
4 63	287	23 0						
4 94	312	29 6						
5 24	341	36 2						
5 52	364	44 3						
5 79	388	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF

Flight No. 25 Run 11
Date 31 MAR 65
Pressure Altitude 130 Ft
Temperature 14.8 °C
Gross Weight 34890 Lb

	LO/TD	50 Ft
V_{gr}	35 7 Kt	48 8 Kt

Wind 2 0 Kt at 80° Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0	0						
–		0						
1 47	57	0						
1 97	81	0						
2 47	111	4						
2 89	137	16						
3 29	163	3 3						
3 65	189	4 9						
3 99	216	9 1						
4 28	237	13 2						
4 58	259	18 2						
4 88	285	24 0						
5 20	310	29 8						
5 50	335	37 4						
5 80	364	50 0						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF LO/TD 50 Ft

Flight No. 25 Run 13
Date 31 MAR 65
Pressure Altitude 130 Ft
Temperature 14 0 °C
Gross Weight 34640 Lb

V_{gr} 35 7 Kt 48 8 Kt
Wind 0 3 Kt at 90°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0	0						
–		0						
154	67	0						
198	92	3						
242	119	8						
279	144	14						
317	171	26						
352	196	50						
386	222	82						
414	246	123						
447	271	173						
478	297	231						
508	321	305						
535	343	378						
567	372	500						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF LO/TD 50 Ft

Flight No. 25 Run 15
Date 31 MAR 65
Pressure Altitude 132 Ft
Temperature 14 7 °C
Gross Weight 34390 Lb

V_{gr} 31 6 Kt 46 6 Kt
Wind 16 Kt at 90°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0	0						
		0						
126	58	0						
184	86	0						
228	113	0						
267	137	4						
303	163	13						
338	188	24						
370	212	53						
402	237	101						
433	261	140						
466	287	214						
497	312	280						
530	339	371						
561	366	500						

XC-142A
USAF S/N 62-5923

TEST TAKEOFF LO/TD 50 Ft

Flight No. 25 Run 17
Date 31 MAR 65
Pressure Altitude 150 Ft
Temperature 15.8 °C
Gross Weight 34150 Lb

V_{gr} 35 8 Kt 49 1 Kt
Wind 12 Kt at 90°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 30 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	0	0						
–		0						
137	52	0						
186	76	0						
223	92	6						
262	122	13						
299	147	23						
337	171	49						
370	195	78						
405	222	134						
436	248	198						
470	274	264						
502	298	346						
534	325	434						
564	353	500						

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Continuous Climb (Climb #1)								
POINT NO.	1	2	3	4	5	6	7	8	9
TIME - SEC	0	10	20	30	40	50	60	110	120
H_{ic} - FT	4459	4670	5120	4875	5882	6735	7580	12038	12780
V_{ic} - KT	1784	1817	1855	1796	1791	1681	495	510	1558
OAT - °C	105	90	76	60	45	30	18	-59	-68
GROSS WEIGHT - LB	—	—	—	—	—	—	—	—	—
TOTAL FUEL FLOW - LB/HR	5641	5290	5228	4962	4695	4585	4466	3936	3937
AVG PROP SPEED - %	920	910	913	918	910	920	913	908	915
AVG TORQUE - %	927	875	843	823	779	747	738	680	677

REMARKS: Flt No 39. Maximum Power
Sea Level Gross Weight 36250 LB

DATE 28 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Continuous Climb (Climb #1)								
POINT NO.	10	11	12	13	14	15	16	17	18
TIME - SEC	135	170	180	230	240	248	312	310	317
H_{ic} - FT	15277	16522	16972	19865	20419	21850	23532	23932	24150
V_{ic} - KT	155.0	149.0	147.4	[illegible]	152.4	150.5	152.4	150.0	149.7
OAT - °C	-7.8	-15.4	-14.6	[illegible]	-21.2	-22.0	-27.2	-28.1	-28.3
GROSS WEIGHT - LB	—	—	—	—	—	—	—	—	—
TOTAL FUEL FLOW - LB/HR	3877	3556	3556	3257	3257	3080	2908	2818	2878
AVG PROP SPEED - %	91.5	91.0	91.0	91.5	91.0	91.3	91.5	90.8	90.5
AVG TORQUE - %	66.2	61.8	60.8	57.3	56.5	56.0	51.2	50.7	51.0

REMARKS:

Flt No 39 Maximum Power

Sea Level Gross Weight 36280 LB

DATE: 28 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Continuous Climb (Climb #1)								
POINT NO.	19	20	21	22					
TIME - SEC	326	336	346	353					
H_{ic} - FT	24178	24825	25023	25116					
V_{ic} - KT	150.0	150.9	150.9	153.8					
OAT - °C	-29.1	-29.8	-30.2	-30.5					
GROSS WEIGHT - LB	—	—	—	—					
TOTAL FUEL FLOW - LB/HR	2859	2801	2783	2763					
AVG PROP SPEED - %	90.4	90.5	90.6	91.0					
AVG TORQUE - %	50.0	49.8	49.7	49.3					

REMARKS:

Flt No 39 Maximum Power

Sea Level Gross Weight 36280 LB

DATE: 28 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Continuous Climb (Climb #2)								
POINT NO.	1	2	3	4	5	6	7	8	9
TIME - SEC	0	14	20	30	40	110	120	188	198
H_{ic} - FT	1495	2290	3187	4140	5389	11587	12284	16668	17548
V_{ic} - KT	171.3	174.6	174.5	169.2	169.0	153.8	153.5	150.7	149.8
OAT - °C	5.2	5.2	7.9	6.5	5.0	-4.0	-6.0	-14.0	-15.3
GROSS WEIGHT - LB	—	—	—	—	—	—	—	—	—
TOTAL FUEL FLOW - LB/HR	5250	52-5	5075	4834	4722	4104	3900	3491	3450
AVG PROP SPEED - %	90.4	90.0	89.0	89.0	89.5	91.6	91.3	91.0	91.0
AVG TORQUE - %	95.1	88.5	85.5	81.0	85.6	68.0	68.0	61.1	60.0

REMARKS:

Flt No 39 Maximum Power

Sea Level Gross Weight 35580 LB

DATE: 28 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Continuous Climb (Climb #2)								
POINT NO.	10	11	12	13	14	15	16	17	18
TIME - SEC	205	250	260	270	332	342	352	362	372
H_{ic} - FT	17765	19951	20352	20818	23339	23735	23932	24330	24750
V_{ic} - KT	149.8	151.8	150.5	152.6	148.0	150.0	149.8	150.0	148.0
OAT - °C	-16.4	-20.7	-21.1	-22.2	-26.8	-27.8	-28.2	-29.7	-27.8
GROSS WEIGHT - LB	—	—	—	—	—	—	—	—	—
TOTAL FUEL FLOW - LB/HR	3425	3219	3180	3150	2963	2887	2844	2814	2755
AVG PROP SPEED - %	91.0	91.0	90.7	91.2	91.5	91.6	90.6	90.4	90.4
AVG TORQUE - %	59.7	56.6	56.2	55.2	50.9	51.1	51.0	50.8	49.3

REMARKS:

Flt No 39 Maximum Power

Sea Level Gross Weight 35580 LB

DATE: 28 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Continuous Climb	(Bag #2)							
POINT NO.	19								
TIME - SEC	382								
H_{ic} - FT	21925								
V_{ic} - KT	150.0								
OAT - °C	-30.2								
GROSS WEIGHT - LB	—								
TOTAL FUEL FLOW - LB/HR	1742								
AVG PROP SPEED - %	106								
AVG TORQUE - %	187								

REMARKS:
Flt No. 39 Maximum Power
Sea Level Gross Weight 35580 LB

DATE: [illegible] April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Maximum Power Level Acceleration								
POINT NO.	1	2	3	4					
TIME - SEC	0	4	8	12					
H_{ic} - FT	1062	1062	1072	1047					
V_{ic} - KT	1858	2028	2177	2310					
OAT - °C	26.0 →								
GROSS WEIGHT - LB	345[illegible] →								
TOTAL FUEL FLOW - LB/HR	—	—	—	—					
AVG PROP SPEED - %	96.5	96.4	96.0	96.1					
AVG TORQUE - %	78.5	79.8	80.7	80.7					

REMARKS:

DATE: 19 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Maximum Power Level Acceleration								
POINT NO.	1	2	3	4	5				
TIME - SEC	0	4	8	12	16				
H_{ic} - FT	1097	1097	1097	1097	1057				
V_{ic} - KT	1755	1978	2150	2270	2295				
OAT - °C	26.0 →								
GROSS WEIGHT - LB	34500 →								
TOTAL FUEL FLOW - LB/HR	—	—	—	—	—				
AVG PROP SPEED - %	96.7	96.3	95.9	95.3	95.9				
AVG TORQUE - %	75.7	79.3	80.1	81.0	81.4				

REMARKS:

DATE: 19 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Maximum Power Level Acceleration								
POINT NO.	1	2	3	4	5				
TIME - SEC	0	4	8	12	16				
H_{ic} - FT	3933	3928	3953	3973	3988				
V_{ic} - KT	1695	1900	2044	2177	2290				
OAT - °C	21.0 →								
GROSS WEIGHT - LB	34950 →								
TOTAL FUEL FLOW - LB/HR	—	—	—	—	—				
AVG PROP SPEED - %	97.2	96.1	96.1	96.1	95.5				
AVG TORQUE - %	78.6	80.5	80.0	80.0	75.6				

REMARKS:

DATE: 19 April 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	MAXIMUM POWER LEVEL ACCELERATION								
POINT NO.	1	2	3	4	5				
TIME - SEC	0	4	8	12	16				
H_{ic} - FT	3973	39[illegible]	3993	4008	3913				
V_{ic} - KT	185 3	200 8	216 5	227 0	237 6				
OAT °C	24 0								
GROSS WEIGHT - LB	34950								
TOTAL FUEL FLOW - LB/HR									
AVG PROP SPEED - %	96 3	96 0	96 0	95 8	95 7				
AVG TORQUE - %	78 8	78 3	79 4	79 9	80 4				

REMARKS:

DATE: 19 APRIL 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	MAXIMUM POWER LEVEL ACCELERATION								
POINT NO.	1	2	3	4	5	6	7	8	
TIME - SEC	0	4	8	12	16	20	24	28	
H_{ic} - FT	8091	8168	8129	8143	8172	8160	8140	8166	
V_{ic} - KT	167 6	[illegible]3	[illegible]	232	229	23[illegible]	231	231 4	
OAT - °C	11.0								
GROSS WEIGHT - LB	35290								
TOTAL FUEL FLOW - LB/HR									
AVG PROP SPEED - %	[illegible]	95 4	95 5	95 0	95 3	95 4	94 9	95 0	
AVG TORQUE - %	71 6	73 8	74 3	75 2	74 8	75.8	75 2	75.2	

REMARKS:

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	MAXIMUM POWER LEVEL ACCELERATION								
POINT NO.	1	2	3	4	5	6	7	8	9
TIME - SEC	0	4	8	12	16	20	24	28	32
H_{ic} - FT	11898	11898	11893	1923	1962	11958	11943	11918	11933
V_{ic} - KT	204 3	210 0	216 0	22 0	223 5	228 5	232 0	235 8	237 2
OAT - °C	7 0								
GROSS WEIGHT - LB	35550								
TOTAL FUEL FLOW - LB/HR									
AVG PROP SPEED - %	95 2	95 3	95 2	95 1	94 9	95 1	95 0	94 9	94 7
AVG TORQUE - %	69 9	69 9	70 5	70 3	70 6	70 5	70 5	70 9	69 5

REMARKS:

DATE: 19 APRIL 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	MAXIMUM POWER LEVEL ACCELERATION								
POINT NO.	1	2	3	4	5	6	7	8	9
TIME - SEC	[illegible]	4	8	12	16	20	24	28	32
H_{ic} - FT	[illegible]	[illegible]	11986	11993	[illegible]2025	2033	[illegible]	[illegible]	[illegible]
V_{ic} - KT	154 8	167 5	178 5	187 9	[illegible]	201 8	[illegible]	[illegible]	[illegible]
OAT - °C	6.0								
GROSS WEIGHT - LB	35550								
TOTAL FUEL FLOW - LB/HR									
AVG PROP SPEED - %	96 9	96[illegible]	96 0	95 8	95 5	95 7	95 6	95 5	95 4
AVG TORQUE - %	6[illegible] 9	70 7	68 6	67 4	66 9	66 8	67 4	[illegible]	68 5

REMARKS:

DATE: 19 APRIL 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST									
POINT NO.	10	11	12	13	14				
TIME - SEC	36	40	44	48	52				
H_{ic} - FT	12013	12033	12028	12015	12015				
V_{ic} - KT	225 5	226 0	230 5	233 9	237 6				
OAT - °C	6 0								
GROSS WEIGHT - LB	35550								
TOTAL FUEL FLOW - LB/HR									
AVG PROP SPEED - %	95 4	95 2	95 2	95 2	95 0				
AVG TORQUE - %	68 4	68 8	68 6	69 3	69 4				

REMARKS:

DATE: [illegible] 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Speed Power								
POINT NO.	1	2	3	4	5	6			
TIME - SEC	0	116	241	440	622	728			
H_{ic} - FT	5920	5960	5975	5890	5960	5850			
V_{ic} - KT	218 8	204 9	190	179 5	160 5	141 0			
OAT - °C	6 4	5 8	5 8	5 5	5 5	5 8			
GROSS WEIGHT - LB	35220	35180	35120	35050	34890	34840			
TOTAL FUEL FLOW - LB/HR	2350	2130	1896	1662	1490	1432			
AVG PROP SPEED - %	79 6	74 0	74 7	75 0	75 6	76 2			
AVG TORQUE - %	89 5	85 0	70 0	59 0	54 0	45 0			
dH/dt - Ft/Sec	3 0	2 C	1 20	0 0	0 0	-2 50			
dV/dt - Ft/Sec²	.0558	0 0	1150	0 0	0 0	3380			

REMARKS:
Flt No 21 Engines #2 and #3 Operating

DATE: 26 Mar 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Speed Power								
POINT NO.	[illegible]	2	3	4	5	6			
TIME - SEC	[illegible]	[illegible]	304	424	508	656			
H_{ic} - FT	100[illegible]	[illegible]10	9918	10040	9978	9978			
V_{ic} - KT	191 0	177 5	169 5	161 5	147 0	132 2			
OAT - °C	1 5	1 2	1 2	0 9	1 8	1 8			
GROSS WEIGHT - LB	33590	33530	33480	33380	33340	33250			
TOTAL FUEL FLOW - LB/HR	1875	1650	1600	1487	1412	1385			
AVG PROP SPEED - %	77 1	76 4	76 2	75 8	75 1	75 9			
AVG TORQUE - %	71 0	61 0	59 0	55 0	50 0	48 0			

REMARKS:
Flt No 21 Engines #2 and #3 Operating

DATE: 26 Mar 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Speed Power								
POINT NO.	1	2	3	4					
TIME - SEC	60	180	252	352					
H_{ic} - FT	20070	20077	20035	19945					
V_{ic} - KT	172 0	158 2	147 0	140 0					
OAT - °C	-30 8	-21 6	-31 0	-30 7					
GROSS WEIGHT - LB	33890	33610	33780	33720					
TOTAL FUEL FLOW - LB/HR	1655	1452	1368	1254					
AVG PROP SPEED - %	78 6	78 2	78 5	78 6					
AVG TORQUE - %	65 0	56 8	53 6	49 5					

REMARKS:
Flt No 21. Engines #2 and #3 Operating.

DATE: 26 Mar 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Speed Power								
POINT NO.	1	2	3	4	5	6			
TIME - SEC	0	206	397	580	691	827			
H_{ic} - FT	11030	10829	10114	10060	10090	9986			
V_{ic} - KT	222.1	204.3	187.4	170.5	165.0	152.4			
OAT - °C	-3.0	2.8	3.2	2.9	3.0	3.2			
GROSS WEIGHT - LB	35005	34930	34860	34635	34585	34480			
TOTAL FUEL FLOW - LB/HR	2412	2159	2278	2144	2028	1967			
AVG PROP SPEED - %	75.4	74.1	75.0	75.6	75.7	75.9			
AVG TORQUE - %	51.2	41.9	36.4	29.8	37.9	36.8			
dh/dt - Ft/Sec	0.0	-2.10	1.25	1.0	.1634	1.0			
dV/dt - Ft/Sec²	0.0	.0845	.0634	0.0	0.0	0.0			

REMARKS: [illegible]
Flt No [illegible] Operating

DATE: 17 Mar 1965

TEST DATA CORRECTED FOR INSTRUMENT ERROR
XC-142A USAF S/N 62-5923

TEST	Speed Power								
POINT NO.	1	2	3	4	5	6			
TIME - SEC	0	157	374	725	847	988			
H_{ic} - FT	25138	25228	25296	25 [illegible]	25205	25418			
V_{ic} - KT	188.3	185.2	179.5	[illegible]7.5	159.0	156.9			
OAT - °C	-32.5	-32.9	33.5	-32.8	-32.8	-32.8			
GROSS WEIGHT - LB	35810	35680	35580	35425	35330	35250			
TOTAL FUEL FLOW - LB/HR	2125	2164	2055	1859	1772	1818			
AVG PROP SPEED - %	75.4	76.4	77.1	76.8	77.6	79.8			
AVG TORQUE - %	45.2	44.3	41.2	35.5	35.5	30.5			
dh/dt - Ft/Sec	0.0	1.20	1.00	1.0	0.87	2.55			
dV/dt - Ft/Sec²	0.0	.30	0.0	0.0	-.0557	0.0			

REMARKS:
Flt No 19 All Eng[illegible] Operating

DATE: 17 Mar 1965

XC-142A
USAF S/N 62-5923

TEST LANDING

Flight No. 23 Run 2
Date 27 MAR 65
Pressure Altitude 245 Ft
Temperature 5.6 °C
Gross Weight 41100 Lb

10/TD 50 Ft
V_{gr} 36.8 Kt 36.2 Kt
Wind 10.7 Kt at 60°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	625	50.0						
81	591	43.3						
124	568	41.5						
175	542	37.6						
218	520	35.2						
270	493	29.4						
320	469	26.2						
369	444	23.7						
412	431	20.4						
466	396	16.4						
515	370	12.2						
563	346	8.2						
600	326	4.9						
656	298	0.0						
705	272							
764	242							
792	223							
829	202							
866	181							
905	162							
953	129							
996	85							
1047	67							
1077	34							
1137	14							
1194	0							

XC-142A
USAF S/N 62-5923

TEST LANDING

Flight No. 24 Run 2
Date 27 MAR 65
Pressure Altitude 230 Ft
Temperature 5.9 °C
Gross Weight 40150 Lb

10/TD 50 Ft
V_{gr} 38.6 Kt 37.7 Kt
Wind 5.5 Kt at 60°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	930	50.0	1434	121				
45	907	49.0	1504	96				
93	882	44.6	1585	71				
140	857	42.3	1694	47				
186	834	38.3	1633	26				
228	811	35.8	1969	0				
274	786	33.?						
320	761	31.8						
369	738	28.5						
409	715	26.1						
458	688	23.6						
500	665	20.4						
560	639	17.1						
588	614	15.5						
[illegible]46	588	13.1						
676	566	11.4						
716	540	9.8						
757	516	8.2						
797	490	4.1						
832	467	3.2						
873	441	1.6						
886	433	0						
908	419							
949	392							
984	371							
1019	343							
1055	319							
1094	296							
1137	269							
1177	246							
1226	224							
1273	192							
1320	170							
1371	148							

XC-142A
USAF S/N 62-5923

TEST LANDING

Flight No. 24 Run 4
Date 27 MAR 65
Pressure Altitude 230 Ft
Temperature 6.2 °C
Gross Weight 40050 Lb

LO/TD 50 Ft
V_{GT} 42.7 Kt 34.3 Kt
Wind 6.3 Kt at 60°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	1016	50.0	1424	86				
45	1005	48.5	1467	164				
91	981	46.8	1514	141				
32	959	44.4	1579	115				
181	932	41.9	1654	88				
225	909	39.4	1717	68				
271	882	35.3	1837	37				
319	859	32.0	1959	19				
367	830	30.4	2146	0				
408	806	28.4						
454	780	26.3						
497	757	23.8						
[illegible]	732	21.5						
585	707	19.7						
624	633	17.2						
666	660	16.4						
712	631	14.0						
750	609	13.1						
793	585	12.3						
834	561	10.4						
870	537	8.2						
918	512	6.6						
957	482	5.7						
992	460	4.9						
1030	438	2.4						
1071	411	0						
1109	385							
1141	362							
1181	336							
1215	311							
1251	288							
1289	262							
1328	240							
1371	214							

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TEST LANDING

Flight No. 24 Run 6
Date 27 MAR 65
Pressure Altitude 235 Ft
Temperature 7.2 °C
Gross Weight 37400 Lb

LO/TD 50 Ft
V_{GT} 41.7 Kt 35.4 Kt
Wind 7.7 Kt at 100°
Degrees from Nose
Wing Incidence 75 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	1230	50.0	1270	182				
36	996	47.2	1320	157				
79	972	43.8	1373	131				
116	951	41.4	1432	108				
158	925	38.8	1502	81				
193	903	36.4	1579	58				
231	880	33.1	1687	32				
275	851	27.8	1798	12				
318	824	26.5	1916	0				
356	800	22.4						
394	776	20.7						
430	752	17.5						
467	729	14.3						
501	706	11.6						
541	679	10.0						
579	650	8.3						
616	627	5.8						
649	606	3.4						
683	581	2.5						
718	557	1.9						
761	527	1.2						
797	501	.5						
829	480	0						
861	453							
895	429							
930	404							
963	380							
999	350							
1035	329							
1070	304							
1125	280							
842	258							
1182	232							
[illegible]	208							

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TEST LANDING

Flight No. 24 Run 8
Date 27 MAR 65
Pressure Altitude 240 Ft
Temperature 8.0 °C
Gross Weight 38850 Lb

LO/TD 50 Ft
V_{GT} 41.1 Kt 37.2 Kt
Wind 6.2 Kt at 100°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	1024	50.0	1259	192				
40	1009	48.6	1302	165				
80	982	46.1	1342	140				
122	956	43.5	1380	119				
160	931	42.7	1427	91				
197	907	40.3	1483	63				
232	884	37.8	1528	45				
266	861	35.2	1598	19				
306	837	34.6	1703	0				
344	811	32.1						
381	787	31.2						
417	763	29.6						
456	739	25.6						
497	711	24.2						
537	687	21.4						
581	661	18.1						
616	636	16.5						
657	609	13.2						
693	586	12.1						
730	562	9.8						
765	539	9.0						
805	513	6.6						
843	488	4.9						
873	462	3.1						
913	439	1.4						
946	415	0						
981	390							
1016	364							
1048	341							
1083	316							
1115	292							
1152	266							
1192	239							
1224	217							

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TEST LANDING

Flight No. [illegible] Run 10
Date 27 MAR 65
Pressure Altitude 240 Ft
Temperature 8.7 °C
Gross Weight 38550 Lb

LO/TD 50 Ft
V_{GT} 41.7 Kt 37.3 Kt
Wind 5.9 Kt at 100°
Degrees from Nose
Wing Incidence [illegible] Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	1166	50.0	1216	331				
35	1151	48.2	1254	306				
71	1128	44.4	1301	282				
108	1102	42.3	1350	259				
144	1070	39.8	1412	236				
179	1042	36.6	1466	209				
219	1025	33.2	1523	185				
255	1000	30.1	1616	152				
290	978	28.2	1674	133				
326	952	25.8	1752	110				
361	929	23.2	1856	81				
395	905	21.6	1948	62				
437	882	17.8	2088	37				
466	851	15.6	2232	18				
501	830	14.2		0				
537	807	[illegible]						
570	781	10.1						
599	751	8.7						
632	730	7.5						
666	708	6.0						
698	683	5.1						
731	660	4.3						
766	632	2.8						
799	608	1.6						
830	585	0						
866	558							
901	531							
937	505							
970	482							
1003	459							
1036	436							
1079	411							
1113	385							
1157	357							

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TEST LANDING

		LO/TD	50 Ft
Flight No. 25	Run 2	V_{gr} 43 1 Kt	40 5 Kt
Date 31 MAR 65		Wind 22 Kt at 90°	
Pressure Altitude 123 Ft		Degrees from Nose	
Temperature 13 2 °C		Wing Incidence 35 Deg	
Gross Weight 41 050 Lb		Flap Position 60 Deg	

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	846	50						
36	827	43 1						
71	802	38 7						
10[illegible]	777	36 4						
151	748	33 0						
182	726	29 7						
216	703	26 6						
[illegible]54	678	23 9						
29[illegible]	652	21 4						
326	629	18 9						
360	606	15 1						
397	580	1 6						
434	557	8 3						
471	528	4 9						
506	503	3 3						
540	475	1 6						
573	453	8						
608	434	0						
641	402							
674	379							
709	363							
75[illegible]	322							
7[illegible]	296							
827	273							
862	251							
901	228							
941	203							
980	180							
1028	155							
1088	12[illegible]							
1146	103							
[illegible]214	79							
1290	53							
1518	0							

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TEST LANDING

		LO/TD	50 Ft
Flight No. 25	Run 4	V_{gr} 37 2 Kt	37 6 Kt
Date 31 MAR 65		Wind 28 Kt at 120°	
Pressure Altitude 125 Ft		Degrees from Nose	
Temperature 13 4 °C		Wing Incidence 35 Deg	
Gross Weight 40 700 Lb		Flap Position 60 Deg	

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	689	50 0						
37	670	47 0						
76	646	40 5						
[illegible]30	612	36 3						
[illegible]74	584	32 2						
211	562	26 4						
249	53[illegible]	23 1						
2[illegible]	513	17 5						
[illegible]28	489	14 1						
368	462	10 7						
406	438	7 4						
444	413	4 8						
480	[illegible]91	2 2						
518	[illegible]64	0						
554	[illegible]41							
591	317							
[illegible]26	293							
667	267							
703	244							
748	216							
788	19[illegible]							
837	163							
889	139							
941	117							
994	94							
1068	68							
1166	38							
1507								
1552	0							

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TEST LANDING

		LO/TD	50 Ft
Flight No. 25	Run 6	V_{gr} 37 8 Kt	33 2 Kt
Date 31 MAR 65		Wind 25 Kt at 100°	
Pressure Altitude 125 Ft		Degrees from Nose	
Temperature 14 0 °C		Wing Incidence 35 Deg	
Gross Weight 37 690 Lb		Flap Position 60 Deg	

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	666	50 0						
38	648	46 2						
80	621	41 3						
124	597	37 2						
166	569	33 9						
205	544	27 8						
246	520	25 0						
2.91	491	21 5						
332	467	18 2						
370	443	13 2						
4.10	419	10 1						
452	392	7.5						
487	369	5.6						
526	342	3.8						
562	321	1.6						
600	294	0						
640	271							
[illegible]61	246							
722	222							
765	197							
812	170							
864	145							
914	122							
985	93							
1481	0							

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TEST LANDING

		LO/TD	50 Ft
Flight No. 25	Run 8	V_{gr} 36.7 Kt	33 7 Kt
Date 31 MAR 65		Wind 19 Kt at 96°	
Pressure Altitude 125 Ft		Degrees from Nose	
Temperature 14 6 °C		Wing Incidence 35 Deg	
Gross Weight 35340 Lb		Flap Position 60 Deg	

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	606	50 0						
44	581	43 1						
88	558	37 8						
125	536	36 5						
165	512	32 4						
211	488	27 3						
256	461	24 1						
297	437	20 7						
339	412	15 9						
382	388	13 2						
423	363	9 6						
467	336	6 9						
509	309	3 3						
546	287	8						
583	264	0						
621	240							
662	213							
708	186							
756	161							
804	138							
859	113							
939	81							
1016	57							
1381								
1441								
14[illegible]1	0							

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TEST LANDING

Flight No. 25 Run 10
Date 31 MAR 65
Pressure Altitude 127 Ft
Temperature 14 4 °C
Gross Weight 34990 Lb

	LO/TD	50 Ft
V_{gr}	36 8 Kt	32 7 Kt

Wind 16 Kt at 90°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	565	50 0						
47	546	45 6						
87	524	41 0						
1 34	499	36 2						
1 77	473	31 3						
2 20	448	26 3						
2 65	423	20 5						
3 09	396	16 5						
3 49	373	11 6						
3 89	350	6 6						
4 30	325	3 3						
4 74	296	0						
5 14	2-1							
5 51	249							
5 87	227							
6 31	202							
6 72	178							
7 22	152							
7 75	127							
8 35	02							
9 05	78							
13 29	0							

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TEST LANDING

Flight No. 25 Run 12
Date 31 MAR 65
Pressure Altitude 130 Ft
Temperature 14 2 °C
Gross Weight 34690 Lb

	LO/TD	50 Ft
V_{gr}	37 5 Kt	36 9 Kt

Wind 15 Kt at 80°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	734	50 0						
42	701	41 1						
80	679	34.6						
1 18	655	30 5						
1 58	629	27 2						
1 97	604	21 4						
2 33	579	18 2						
2 79	553	14 0						
3 20	527	11 5						
3 57	502	9.1						
3 94	480	7 4						
4 34	453	5 8						
4 77	427	4 3						
5 14	403	3 0						
5 49	377	[illegible]						
5 86	35[illegible]	9						
6 22	32[illegible]	0						
6 55	305							
6 93	278							
7 38	253							
7 76	230							
8 22	205							
8 72	178							
9 20	156							
9 75	131							
10 41	107							
11 19	80							
11 99	56							
15.51								
16.04	0							

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TEST LANDING

Flight No. 25 Run 14
Date 31 MAR 65
Pressure Altitude 130 Ft
Temperature 14 4 °C
Gross Weight 34590 Lb

	LO/TD	50 Ft
V_{gr}	36 8 Kt	34 1 Kt

Wind 0 Kt at 0°
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	556	50 0						
40	529	43.1						
.82	505	39 7						
1 26	479	33 9						
1 70	454	29 8						
2 10	430	26 5						
2 53	404	20 1						
2 93	378	16.5						
3 32	355	13 2						
3 72	331	9 7						
4 16	304	5 8						
4 56	279	2 4						
4 94	255	0						
5 34	230							
5 72	207							
6.13	18[illegible]							
6 62	158							
7 08	134							
7 68	107							
8 26	83							
9 10	56							
13 19								
13 60	0							

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TEST LANDING

Flight No. 25 Run 16
Date 31 MAR 65
Pressure Altitude 140 Ft
Temperature 14 6 °C
Gross Weight 34340 Lb

	LO/TD	50 Ft
V_{gr}	34 7 Kt	33 2 Kt

Wind 0 Kt at 0
Degrees from Nose
Wing Incidence 35 Deg
Flap Position 60 Deg

Time	Distance	Height	Time	Distance	Height	Time	Distance	Height
0	534	50 0						
48	495	42 1						
90	471	37 1						
1 34	447	31 4						
1 80	421	27 1						
2 24	396	22 2						
2.65	373	17 3						
3 07	349	13 2						
3 50	322	9.1						
3.89	300	6.5						
4 33	273	3.2						
4 80	246	0						
5 23	218							
5 59	197							
5 98	175							
[illegible] 50	148							
6 99	124							
7 52	101							
8 23	75							
8 98	51							
12 74								
13 32	0							

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